THE COLLEGE OF AERONAUTICS

CRANFIELD

AEROPLANE DESIGN STUDIES
MACH 2.2 AND MACH 3.0 SUPERSONIC AIRLINERS
(ACADEMIC YEARS 1960 AND 1962)

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SUMMARY

This report is divided into three parts. The first two of these describe the A-60, Mach 2.2 airliner and the A-62, Mach 3.0 airliner design studies respectively. Apart from the different cruise speeds these two aircraft were designed to meet the same basic requirements and the third part of the report is a comparison of them.

The Mach 2.2 design was based upon the use of a slender, integrated, delta layout with six turbojet engines buried in the rear fuselage. It was intended to carry up to 120 passengers over transatlantic ranges. Although the chosen engine installation enabled a compact aircraft to be designed it did introduce severe structural and installation difficulties.

A canard delta arrangement was proposed for the Mach 3.0 aircraft. Drooping of the wing tips for supersonic flight was found to confer important stability advantages without introducing an unacceptable weight penalty. The steel structure was designed around the use of both corrugated reinforced and honeycomb sandwich skins, the former being preferable. An interesting feature was the choice of a sealed, cryogenic, environmental control system. This was found to be very attractive but as it proved to be somewhat heavier than anticipated it is suggested that a good compromise could be obtained by using a more conventional system for subsonic flight phases.

The major conclusion from the comparison between the two study aircraft was that in many respects there is very little to choose between them. However the Mach 2.2 aircraft represents a more logical step from existing airliner designs and presents fewer materials problems. As it is comparable economically it represents a better choice for a first generation supersonic design.

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Introduction

The formation of the Supersonic Transport Advisory Committee in 1957 marked the beginning of interest in supersonic airliners at the College of Aeronautics. Early investigations indicated that the use of a delta winged aircraft flying at just under twice the speed of sound was a promising possibility. The salient design features of this type of aircraft were investigated and initial project studies undertaken. By 1960 sufficient background information had been accumulated to enable the work to be extended by using it as the basis for a student project study [(1) (2)] in the Department of Aircraft Design. Known as the Project A-60 the design was investigated in some detail by a team of fourteen students during the 1960-61 academic year. The individual allocation of components is given in Appendix A. The aircraft was designed to cruise at a Mach number of 2.2 and carry up to 120 passengers over transatlantic routes.

Although the choice of Mach 2.2 for the cruising speed coincided with Anglo-French thoughts on the subject elsewhere considerable interest had been shown in aircraft designed to fly at higher speeds. In view of this it was decided to supplement the work carried out on the A-60 design by undertaking a similar study of an airliner designed to cruise at three times the speed of sound, this being known as the A-62. In order that a direct comparison of the two types could be made the only difference in the basic requirements was the choice of cruising speed. Appendix B gives the allocation of the components of the later design amongst the fifteen students who worked on it during the 1962-63 academic year.

Figures 1 and 2 are photographs of models of the A-60 and A-62 designs, respectively. Typical flight programmes for the two aircraft are shown in Figures 3 and 4.

PART 1

MACH 2.2 AIRLINER, PROJECT A-60

1.0 Preliminary Investigations

The initial work which lead eventually to the A-60 design was carried out by Spillman [3], who demonstrated the importance of achieving a minimum total volume for a supersonic aircraft. Following from the work of Küchmann [4] and others at the Royal Aircraft Establishment he proposed a layout having modified delta wing of "ogee" planform integrated with the fuselage. The resulting design is illustrated in Figure 5. With an estimated take off weight of 310,000 lb., the aircraft was intended to carry 100 passengers over transatlantic routes at a cruise Mach number of 1.8. Power was provided by eight unspecified turbojet engines of 12,000 lb. thrust mounted in the rear fuselage. The wing leading edge camber was designed for a cruise lift coefficient of 0.1. Certain of the layout problems associated with this initial design were investigated by Capey [5]. In particular he suggested that the overall volume could be minimised by having six abreast passenger seating in a cabin of horizontal "double-bubble" cross section. Shown in Figure 6, this arrangement incorporated a double width gangway divided by a lengthwise bulkhead which served to react the pressure shell discontinuity loads. The external aerodynamic shape was maintained by an envelope structure used to transmit wing bending loads across the fuselage. A one forty-eighth scale model of the design was tested at low speed in the Aerodynamics Department of the College and a number of undesirable features became apparent. The sharp lower edges of the fuselage surface caused severe flow separation to occur and the aircraft was found to be deficient in both longitudinal and directional stability. A more critical survey of the layout indicated the possibility of significant structural problems. For example the chosen area distribution was only achieved by having very thin wing tips which gave insufficient depth for control hinges, and the undercarriage stowage aft of the cabin introduced serious structural discontinuities.

1.1 The A-60 Design

A completely new design was prepared as the result of this experience and it became known as the project A-60. The overall layout of the earlier design was retained, as is shown in Figure 7 which is a general arrangement drawing of the A-60. A cruise Mach number of 2.2 was chosen as being the maximum likely to be possible with a structure composed mainly of light alloys. The wing design, however, enables useful subsonic leading edge flow to be maintained up to approximately Mach 2.35. The integrated tail-less slender wing layout is based on a 73° delta, the leading edge of which is curved at both the root and tip to improve the spanwise lift distribution. The gross area of 5,500 sq. ft. and span of 77 ft. represent a compromise between lowspeed and cruise requirements. A design payload of 22,700 lb., 108 passengers, can be carried over a range of 3,250 n. miles with fuel reserves for 200 n. miles diversion and a total standoff time of one hour. The full payload of 120 passengers can be carried over a somewhat reduced range. With maximum take off and landing weights of 325,000 lb.

and 190,000 lb. respectively the lift off and approach speeds are 200 knots and 150 knots. The former is determined by the elevator power necessary to lift the nose and the latter by the maximum usable lift coefficient of 0.57 which results from an incidence limitation of 15°. The aircraft uses 10,000 ft. of runway during take off.

The cross section area distribution was determined by the minimum area at the pilot's position, fuel stowage volume and the minimum acceptable structure depth in the region of the trailing edge. Maximum area occurs at 60% of the length aft of the nose and with a constant cabin cross section the wing sections follow automatically. These are basically biconvex in shape and vary in thickness from root to tip as shown in Figure 8. The cruise is thrust limited and the lift coefficient of 0.096 corresponds to a lift-drag ratio of approximately 8.5. The maximum lift-drag ratio of 9.0 occurs when the lift coefficient is 0.134. The wing leading edge camber was based on a design lift coefficient of 0.05.

The six, 18,000 lb. sea level static thrust Bristol Siddeley Olympus 591 engines are mounted in two rows in the rear fuselage. The two dimensional wedge intakes are located at the end of a ramp on the upper surface of the fuselage. The fin structure passes round the upper centre engine jet pipe and much of the fin root is cut away for engine access. A considerable design improvement would be achieved if the six engines were to be replaced by four of the later and more powerful Olympus 593 variants. Not only would the structure be lighter and engine accessibility be much improved, but 120 passengers could be carried over 3,500 n. miles range without increase of take off weight. All the fuel is carried in integral wing tanks.

A somewhat unusual cabin layout has been adopted. The basic aircraft configuration presents certain difficulties in that nearly all of the volume available for payload is forward of the centre of gravity and access to the rear of this region can only be made from below. As shown in Figures 9 and 10 the main cabin is unobstructed over its whole length and six abreast seating is used. The wasteful and undesirable double gangway of the initial design is eliminated by recourse to a "treble bubble" cross section shape. This is illustrated in Figure 6 and as can be seen compares favourably with an elliptical shape from the point of view of cross sectional area. The pressure discontinuity loads are reacted by a series of vertical posts which coincide with every other seat and are removable with the seats when cabin layout changes are necessary. A seat pitch of 33 inches is required to enable 120 passengers to be carried, but this can be increased to 40 inches for 108 passengers. The outer fuselage shape is approximately elliptical but blends into the wing over the greater part of its length. The provision of deep root ribs along the sides of the cabin results in a double walled structure which is beneficial both from safety and insulation requirements. The integrated layout precludes the use of windows except at the extreme rear of the cabin, but roof lights and escape hatches are provided. Forward of the main passenger compartment the pressure shell discontinuity loads are reacted by full depth bulkheads, the nosewheel retracting into the space between them. The crew cabin has a basically circular cross section and the transition into this shape is achieved by bringing the bulkheads together on the centreline. The length of the cabin where the full depth bulkheads occur is used for freight and baggage holds, a pantry and four toilets. The main passenger entry door is also located in this region. Some stowage space is also available over the nosewheel bay and additional freight could be stowed in the deep root leading edge.

2.0 Detail Specification of the A-60

A detail specification of the A-60 design is given in Appendix C. Together with the weight breakdown to be found in Table 1 and certain load distributions this appendix represents the initial information given to the students. Table 1 includes both predicted weights and those estimated as a result of the detail design work. As far as possible a direct comparison of the structural weight breakdown has been made, but minor descrepancies are inevitable in view of the complex structure.

3.0 Description of the Structure of the A-60

The structure of the project A-60 was designed to enable a life of 30,000 hours to be achieved. Of this time, 20,000 hours were assumed to be spent at the cruising speed when the greater part of the structure would be subjected to temperatures of the order of 120°C to 130°C, dependent upon the surface finish. The use of light alloy in this environment introduces a significant creep problem and after a survey of possible materials had been carried out RR58 alloy (D.T.D. 5070) was chosen as the most promising material. Limitation of the creep strain to 0.1% during the life of the aircraft lead to the fixing of the working stress level at 14,000 p.s.i., which corresponds to 1.2g normal acceleration where this is applicable. On this basis the stress arising at proof loading was found to be of the order of 30,000 p.s.i. in most cases. In those parts of the structure not subjected to kinetic heating, as for example the inner cabin, L73 alloy was preferred. The undercarriage was largely designed in S99 steel and stainless steel was used in the intake design, titanium only occurring in the engine firewalls.

The disposition of the main structural members is shown in Figure 11. Conventional skin-stringer construction is used whereever possible but the rear portion of the wing and fin use machined integral skins, and honeycomb sandwich construction was found to be desirable over part of the forward wing. The integrated wing-fuselage construction gives rise to an assembly problem due to the large size of the basic component.[6] As the lift is developed over the greater part of the planform and it is balanced by local inertia forces, the airloading does not normally give rise to large shear forces and bending moments. The main exceptions to this are the loads arising from control deflection, particularly those due to the aileron, which give critical spanwise cases over the rear portion of the wing. The largest concentrated loads occur during landing and, in spite of the low proof reaction factor of 1.5, these give longitudinal bending moments which are an order of magnitude greater than those which result from the airloading. A one twentieth scale dynamic model was constructed [7] to investigate this and associated problems [8] [9]. Manufactured in wood, this model is illustrated in Figure 17.

3.1 Wing-Fuselage Component

Although the wing-fuselage part of the structure is ideally one constructional unit it is convenient to describe it as a number of smaller items. The three point landing case gives a maximum unfactored bending moment of approximately 6×10^6 lb. ft, whilst laterally the maximum figure of 1.8×10^6 lb. ft. arises when the rudder is instantaneously deflected at high subsonic speed.

Nose Fuselage

Apart from the crew floor, windscreen and radome, the nose fuselage is constructed entirely of D.T.D. 5070 sheet. Frames placed at a nominal pitch of 12 inches support 20G Zed stringers which are located round the section at an average pitch of 5.4 inches.Forward of Station 135.4 the skinning is 20G and aft of this section it is 18G. Spin dimpling is used throughout and 18G crack stopping bands are located at each frame, outside the stringers. Machined angle section longerons are used as boundary members for the canopy cutout. The forward end of the crew cabin is closed by a 24G membrane bulkhead and the nose shape is completed by a phenolic resin glass fibre radome. Honeycomb sandwich construction is used for the crew floor which is supported from the fuselage frames by lateral I section beams placed at 36 inches pitch, with additional longitudinal beams at 12 inches pitch. The windscreen structure consists of four triangular RR 58 forgings bolted together to give double pillars at the edges of the panels. The windscreen itself is of composite construction. Air, pressure and thermal loading are reacted by the outer layer of 0.6 inches thick toughened soda-lime glass which is subjected to a maximum stress of 4500 p.s.i. The inner wall is a laminated glass-vynal bird proof screen which has cooling air passed over it to prevent the temperature exceeding 80°C.

Main Cabin

The internal cabin reacts the normal pressure differential of 10.5 p.s.i., and as it is rigidly connected to the outer shell it also shares in carrying the other loading. In 1 g flight the maximum tensile stress developed in it is 12,500 p.s.i. Should this internal cabin fail for any reason the outer shell is designed to be able to withstand a pressure differential of 5.25 p.s.i. The outer shell, which is really the main fuselage structure, uses 18G Zed stringers placed at 3.5 inches pitch to reinforce the skin. This varies in thickness from 12G to 20G along the cabin length. Frames located at approximately 18 inches pitch support the inner cabin as is shown in Figure 12. Additional, intermediate frames are located round the 20G thick skin of the inner cabin. Vertical ties support the four angle section longerons which are located at the discontinuities in the cabin wall. Also shown in Figure 12 is the substantial frame used to transmit the main undercarriage loads into the structure. This is built up of L65 forgings and causes discontinuities to be introduced into the stringers and longerons. The design undercarriage case gives rise to a maximum stress of 18,000 p.s.i. in the cabin. A flat, built up, pressure bulkhead seals the rear end of the cabin proper, whilst at the forward end there are two further bulkheads. These two both use honeycomb sandwich construction, the extreme forward one serving as a pressure seal between the crew and passenger compartments, and the other as a mounting for the nosewheel. The cabin floor is in longitudinal sections which are supported by transverse beams attached to the outer shell.

Wing

The basic wing structure consists of some 34 spanwise members and 12 main ribs on each side of the aircraft. The deep root ribs complete the sides of the outer fuselage shell. The critical spanwise bending moment of 2×10^6 lb. ft. arises during maximum normal acceleration with pitching acceleration at the supersonic design diving speed. Maximum spanwise shear force occurs approximately mid-way along the span when the ailerons are fully deflected at high speed and it amounts to 1.7×10^5 lb. Gust cases were not found to be critical, although this is only marginally true in the final standoff part of reserve flight. The complex structure was analysed by matrix methods (10) and the technique has subsequently been developed on a more general basis (11) (12).

The spars are placed normal to the aircraft centreline and are natural extensions of alternate fuselage frames. To facilitate the support of the aileron hinges the five rearmost ones are swept back from a point about two thirds of the way out along the span. The pitch of the main ribs is approximately 30 inches with subsidiary members located between them in the rear part of the planform. Conventional skin-stringer construction is used over a large region of the wing with Zed stringers running spanwise at a mean pitch of 4.0 inches. The fuel tanks in the forward root region are relatively deep and pressure can rise to approximately 20 p.s.i. However other loading is relatively low in this region and it is expedient to use honeycomb sandwich skin panels since this has the additional merit of reducing the weight of tank insulation and enables a good surface finish to be achieved. The 0.8 inch deep "Aeroweb" honeycomb is "Hidux" bonded to 22G facings of D.T.D. 5070. Panel joints are made by means of Tee and X extrusions in RR 58. Forward of the tank region the 18G D.T.D.5070 skins are reinforced by 18G Zed stringers and supported by fluted 20G rib and spar webs. The leading edge uses closely spaced 20G riblets mounted normally on an 18G web. Honeycomb sandwich panels are also used for the flat, deep root ribs which have Tee section extruded upper and lower booms. These members are tank walls and react some 25% of the fore and aft bending load.

The rear portion of the wing is subjected to substantial spanwise loading, the thin outer tip being a special problem due to the presence of an aileron hinge. The last five spars are part of an integrally machined box structure in which the skin thickness varies from 0.08 inches outboard to 0.15 inches inboard. Outboard of the point at which the spars are kinked the ribs are more highly loaded than the spars and the latter are therefore discontinued at the intersections of the two. Spars and ribs generally use single plate webs with back to back angles for the booms. Inspection of the integral fuel tanks is a serious problem in such a shallow, complex structure. Removable panels are provided in the lower skins where necessary and these are supplemented by holes in spar and rib webs. The trailing edge of the wing aft of the rearmost spar is built as a series of hinged panels to give access to the control hinges and operating mechanisms.

Elevators

The maximum unfactored elevator load is 74,000 ob. (total) and it occurs at maximum normal acceleration with zero pitching acceleration in supersonic cruise. The elevators are comparatively deep and the construction is based upon a plate spar and plate ribs placed at a mean pitch of 5.5 inches. Six spanwise Zed stringers further reduce the size of the 18G skin panels and prevent the possibility of panel flutter. Each of the elevators is operated by three pairs of hydraulic jacks which are located at the lower surface in fairings. There is no mechanical connection port to starboard.

Ailerons

The maximum load of 39,000 lb. on each surface occurs when the ailerons are deflected to prevent the aircraft rolling during yawing at high subsonic speeds. Compared with the elevators the outboard ends of the ailerons are very shallow and "Hidux" bonded full depth "Aeroweb" honeycomb is used to ensure adequate stiffness. Inboard the skins are supported by six spanwise stringers and ribs placed at 5 inches pitch. The thickness of the skin forward of the plate spar is 16G and aft of it the thickness is 20G. Each aileron is operated by four pairs of jacks which are located in fairings below the aerofoil surface.

Rear Fuselage

The construction of the fuselage aft of the main pressure cabin is complicated by the way in which the powerplant installation interferes with the input of loads from the wing and fin. The seven rearmost wing spars and ths skins between them pass through the fuselage below the engine installation, but in order to do this they are cranked and reduced in depth. The centre lengths of these spars, between the two sides of the fuselage, are machined light alloy forgings, production joints being incorporated at the points where they join the outer spars. Access panels are provided in this centre wing box to facilitate installation and servicing of the lower engine accessories. The spar at Station 31 is full depth across the fuselage and acts as the standby pressure bulkhead. It has vertical I and horizontal Zed section stiffeners.

The fin loads are taken into the rear fuselage on the six aft fuselage frames. Of these the three forward ones coincide with the last three wing pickup frames. Since the fin is mounted astride the centre engine it is necessary to take the fin spar frames round the jet pipe. The resulting frame design is complex and uses a forging as a centre arch mounted on a lower rectangular section which is designed to transmit the loads past the outer engines to the fuselage sides. Zed section stringers are used to reinforce the outer skins which vary in thickness from 22G at the rear to 14G just behind the cabin. A substantial portion of the upper part of the fuselage structure consists of full depth honeycomb access doors for the top three powerplants. The exhaust nozzle assembly is mounted off the aftmost fuselage frame and is removed to enable the lower three engines to be withdrawn rearwards. The engine main mounting trunnions are supported off the body sides and two inner ribs of braced construction with titanium faces so that they serve as firewalls. The structure would be considerably simplified if only four engines were installed, as mentioned in paragraph 1.1, since the fin spars would then be able to pass between the pairs of engines.

3.2 Fin and Rudder

Instantaneous rudder deflection at high subsonic speed gives rise to the maximum fin load of 95,000 lb., unfactored. The maximum rudder design load is 32,000 lb. and this occurs in equilibrium yawed flight at supersonic speed. The fin structure consists of three distinct parts. The main load carrying box is located at the rear of the surface and uses six spars which coincide at the lower end with rear fuselage frames. Each of these spars is swept back at 25.5° and is constructed from plate webs and extruded Tee section booms. The skin is tapered from 0.1 inches thickness at the root to 0.05 inches thickness at the tip and it is reinforced by Zed section spanwise stringers at 3.5 inches pitch. The chordwise plate ribs have a mean pitch of 18 inches. At the root the skin and stringer loads diffuse into the spar frames, the skin terminating at a root rib. The centre portion of the structure has no root attachment as it is necessary to provide a large cutout for the removal of the upper centre engine. The loads in this section are mostly transmitted aft to the main box but some support is forthcoming from three short spars at the root leading edge.

The rudder is mounted off the fin rear spar at four hinge points. It is actuated by two pairs of jacks located at the extreme hinges. The lower of these pairs is located in the top of the fuselage, and the upper are housed in fairings on the sides of the fin. The small depth of the rudder section and the noise environment due to the proximity of the power plants led to the choice of honeycomb sandwich construction for the skin panels. These are supported by plate ribs.

3.3 Undercarriage

The main undercarriage employs an eight tyred bogie arrangement which enables an L.C.N. of 80 to be achieved at the all up weight of the aircraft. The maximum unfactored loads on a single main undercarriage unit are 148,300 lb. vertical and 118,000 lb. drag during braked taxi-ing, and 31,000 lb. side in a normal landing.

The liquid spring shock absorber is separate from the main leg which houses it. It has a maximum reaction factor of 1.5 at the proof vertical descent velocity of 12 ft./sec. The effect of temperature variation on the shock absorber is minimised by using "MS 200/20" silicone fluid with a light aromatic oil additive and cooling the undercarriage bay, which is located in the wing root. A recuperator is used to maintain the unit charging pressure at 2000 p.s.i. Both the main leg and bogie beam are constructed in S99 steel, the latter being a hollow circular forging. The single support strut has a knuckle joint and folds to allow the undercarriage unit to retract forwards into the wing. The bogie is trimmed to lie in line with the leg by a combined hop damper, trimming jack unit. Provision is made to lock the bogie on the trimming jack when the unit is extended to increase the effective wheelbase. This is necessary to give adequate ground stability when the aircraft is empty. The multiplate disc brakes are cooled by electrically driven fans to reduce the turn round time when this is limited by brake heat dissipation. Most of the components are interchangeable port to starboard, the main exception being the leg itself. This difficulty can be overcome if the duplication of the strut and toggle attachment lugs on both sides is accepted.

The nose undercarriage also uses a telescopic liquid spring shock absorber. The maximum unfactored loads are 71,750 lb. vertically and 28,700 lb. drag during dynamic braking, and 10,950 lb. side load in a normal three point landing. Steering is from the top of the leg, which is supported by two struts. These struts are articulated and fold to enable the unit to retract aft into the fuselage bay. The relatively large diameter, thin, twin wheels mounted on a live axle are necessary to enable the unit to be housed between the longitudinal pressure cabin bulkheads.

4.0 Project A-60 Installations and Systems

The locations of the major components of the systems and installations of the Project A-60 can be seen in Figures 9 and 10.

4.1 Engine Installation and Air Intakes

The location of the six Olympus 591 engines in the rear fuselage presented serious structural and installation problems. Although the layout of two rows of three engines enables a compact arrangement to be achieved, the accessibility of the centre pair of engines is particularly poor and is aggravated by the presence of the fin structure above and the spanwise wing box below them. The solution adopted was the location of the upper engines forward of the main fin structure and the acceptance of a large cutout in the fin root to enable the centre engine to be removed. This engine is first lifted vertically and then outwards at an angle of approximately 45° to the vertical. The outer two of the upper engines are removed vertically, special lifting beams being necessary due to the awkward layout and the height of some 23 feet above the ground. It is proposed that these lifting beams should be mounted off the fin and fuselage structure. The lower row of engines can only be removed in an aft direction, and this requires the dismantling of the exhaust nozzle assembly and removal of the jet pipes. The latter are mounted from overhead monorails to facilitate this operation. Mounting of the individual engines is conventional in that two main trunnions and a front suspension are used. In each case the port trunnion is fixed laterally to enable side load to be reacted, the starboard one being free to slide to cater for diametrical expansion. Longitudinal expansion is allowed to occur by arranging the front suspensions with swinging links capable only of transmitting vertical loads. The lower engines each have a single front suspension whilst the upper engines are supported on either side of the intake casing. In the case of the lower engines, the inner trunnions are not readily accessible, and a remotely controlled, semi-automatic, locking device similar to a bomb release is employed.

The jet pipes are designed as a double walled construction. An insulating blanket of "Refrasil" is laid over the stainless steel inner pipe and a light alloy outer pipe is placed around this to leave a 3 inches deep annulus through which cooling air is passed. The variable area two dimensional convergent nozzles are mounted off the rear face of the fuselage structure. The double row arrangement precludes the use of thrust reverses. Titanium firewalls, placed both horizontally between the two rows of engines and incorporated in the vertical mounting ribs, isolate each engine. Provision is made for injecting fire extinguishant into each of these bays, sufficient being installed for two-shot operation. As many as possible of the upper engine accessories are located at the top of the bay, whilst those on the lower engines are mounted off the bottom of the powerplants, access being obtained through removable panels in the wing box structure.

The three shock air intakes are rectangular in section, the geometry being varied by means of a horizontal moving ramp, as shown in Figure 13. Upper fuselage boundary layer air is bled off from below the intake proper and used to ventilate the engine bay before it is extracted at the exhaust nozzles. Bleed doors placed in the intake walls aft of the variable entry assembly serve two main purposes. They are used to spill small quantities of surplus air, thereby assisting in intake control, and to dump large quantities of air when an engine is shut down in flight. The doors in the upper three and two outer lower intakes are located in the outer walls of the rear fuselage structure, but the spill air from the centre lower intake has to be fed into a duct which passes through the boundary layer, is split and eventually ends at the fuselage side.

The lips of the intakes are manufactured from hollow stainless steel extrusions through which cooling air is passed. The panels of the moving wedge portion use a honeycomb sandwich construction with 20G D.T.D. 5070 facings and a "Hidux" bonded, "Aeroweb H" core. A more conventional construction is employed for the main walls where the 22G skins are supported by Zed stringers placed at 2.75 inches pitch and closely spaced channel section formers. Sandwich panels are also used for the flat bleed doors. The intake geometry is varied by a remotely located hydraulic motor system which operates through a worm drive and recirculating ball screw jacks. Although the maximum normal intake pressure of 13 p.s.i. occurs during ground running, an ultimate factor of three has been used to cover the case of compressor surge.

4.2 Power Supplies

All the auxiliary power is derived from the main engines, but an alternative scheme using an A.P.U. has been investigated [13]. The three basic power supply requirements are:-

1. 4 lb. of air per second for cooling and cabin conditioning.
2. Hydraulic power amounting to a maximum equivalent of 400 H.P. for the flying controls, air intake and undercarriage.
3. Aircraft electrical services totalling approximately 150 KVA.

The actual design makes no provision for deicing but the original weight prediction included an allowance for this contingency.

Each of the six engines is fitted with a split mechanical-pneumatic constant speed drive of the type developed by Plessey. This is also used for engine starting and ground running from an external compressed air supply, and in emergency can be driven directly from intake air should an engine failure occur in flight. The constant speed drive units are each assumed to be capable of a normal output of 200 H.P. and it has been estimated that the engine bleed required for this should not exceed 6 lb. of air per second. Provision is therefore made to tap up to 7.5 lb. of air per second from each engine, the additional 1.5 lb. being allocated for the direct cabin supply. Only three engines are actually required to supply this quantity at any one time. The tapping is at the low pressure stage of the compressor, but provision is made for change-over to a high pressure tapping during idling conditions. The constant speed units on the lower engines each drive a group of three hydraulic pumps which give an estimated total equivalent output of about 480 H.P. This enables the full load to be carried with two pumps inoperative and all normal flight requirements to be maintained with four inoperative. Each of the constant speed units on the upper engines drives a 100 KVA alternator, two of these in parallel being used normally, with the third as a standby. One alternator is sufficient to supply all essential services.

4.3 Flying Control System

A fully powered flying control system is used. In view of the relative expansion problem created by the cruise temperature environment it was decided to use a quadruplicated electrical signalling system. Three pairs of jacks are provided on each elevator and four pairs on each aileron. The rudder has two pairs. Two separate feed systems are used, one to each of the jacks in a pair. Although all the jacks are normally used simultaneously the design makes provision for full operation in the event of a failure of either a single or complete pair of jacks. Normal essential operation can be maintained by one set of jacks should there be a failure in the feed system.

4.4 Fuel System

The fuel system, which has been designed to use AVTUR, has a total capacity of 20,300 imperial gallons. One half of the complete system is shown in Figure 14. Although a cross feed is provided each half of the system is independent and supplies fuel to three engines. The fuel is contained in seven tanks in each wing, and of these five are connected to the engines through a proportioning system. Engine bleed compressed air is used to drive the fuel proportioner. Of the other two tanks, number one holds reserve fuel and number five is used during the climb. The system uses booster pump feed, the pumps being located in sumps placed at the inboard end of each tank. The pumps in the tanks connected through the proportioner are duplicated but those in tanks one and five are triplicated. Clack valves located at the front, centre and rear of each tank keep the sumps full of fuel independent of the attitude of the aircraft. A nitrogen-air inward venting system is provided. Nitrogen is stored in liquid form in the tank region and is mixed with air bled from the engines. This mixing is used to economise on the quantity of nitrogen required, and the gas pressure is maintained at 1 p.s.i. above ambient. Outward venting is by valves located in the upper four corners of each tank which lead into two interconnected gallery systems.

Refuelling is carried out at two points, one on each of the undercarriage legs. Using two 500 gallon per minute capacity bowsers the aircraft can be refuelled in 27 minutes. Fuel can be jettisoned through pipes located at the trailing edge of the wing between the ailerons and elevators. The tanks are integral with the structure which is wet assembled for sealing with fillets added subsequently at all joints. In those tanks where the construction does not use honeycomb sandwich skinning it is necessary to provide insulation and for this purpose the use of urethane plastic is suggested. The temperature of the tank fuel rises to a maximum of 90°C during a normal cruise. The fuel in the feed lines reaches 100°C.

4.5 Cabin Air Conditioning System

The cabin air conditioning system is designed to supply 1.2 lb. of air per minute to each passenger in a cabin where the pressure does not fall below that at 6,000 ft. altitude. Cabin minimum pressure is determined by the maximum tolerable cabin rate of descent at the end of cruise. During engine idling and in emergency a reduced air supply of 0.5 lb. per minute to each passenger has been accepted. As is shown in the schematic diagram in Figure 15 the air is obtained by tapping the low pressure engine compressor at 29 p.s.i. Three stages of cooling are employed before the air is distributed in the cabin. The first of these uses heat exchangers placed in the intake boundary layer air bleed and these reduce the temperature from 295°C to 173°C. Heat exchangers are also placed in the fuel lines to the engines and these enable a further temperature drop of 80°C to be obtained. Finally a Freon 11 refrigerator system lowers the air temperature to 8°C, surplus heat being dumped into the fuel tanks. At this stage of the process the total fresh air supply of 160 lb. per minute is mixed with 240 lb. per minute of recirculated air, which has been extracted from the main cabin. Air removed from the crew compartment is used to cool electronic equipment and that taken from the toilets and galley is fed into the wheel bays before it is dumped overboard. In the cooling process approximately 56 tons of heat are transmitted to the fuel lines by the fresh air and a further 48 tons to the fuel tanks by the refrigerator system. The whole system is duplicated, and additional duplication is incorporated in the refrigeration and recirculation sections.

The flow of air into the passenger cabin is illustrated in Figure 16. Use is made of integral ducting round the cabin walls to provide some insulation, this being supplemented by a 2.5 inches thick layer of fibreglass placed on the inside of the inner cabin skin. The air gap between the inner and outer shells has a big effect in reducing the weight of insulation which is required. The ducting and fibreglass insulation has been estimated to weigh 1,400 lb.

5.0 Special Features of the A-60 Design

The two most interesting features investigated in the design are the integrated layout with the separate internal cabin and the rear engine installation.

5.1 The Integrated Layout with Internal Cabin

Structurally the integrated layout is advantageous in that it gives a deep wing root over a large part of the chord, although the depth is somewhat embarrassing near the leading edge where the loading intensity is low. Stowage of the undercarriage is straightforward and a large number of the fuel tanks have an appreciable depth. Against this must be set the impossibility of having normal windows and, possibly more important, the difficulty of providing access doors at the rear of the cabin without a substantial loss of volume. Structurally no unusual problems were encountered and since ingenuity with internal decor should enable the lack of windows to be overcome it would seem to be logical to use the integrated layout on a large slender aircraft.

Although the integrated layout lends itself to the incorporation of a double walled cabin, this is not an essential feature of the arrangement. Attempts made to mount the internal cabin independently of the main structure proved to be in vain and it was therefore necessary to design it to withstand normal bending loads. However it did not contribute substantially in reducing the material required in the outer shell and the main advantages which can be claimed for it are the extra margin of safety, which in a correct design is really only psychological, and the reduction of insulation. A direct weight penalty of some 2,400 lb. is incurred and the saving in weight of the insulation and outer shell is not likely to offset this. The "treble bubble" layout does give a compact cabin which makes full use of the available volume.

5.2 The Buried Rear Engine Installation

The location of the six powerplants in the rear of the fuselage resulted in a compact aircraft but introduced most of the problems encountered in the structural design. Many of these problems were directly the result of using six engines since this inevitably implies that the centre two are almost inaccessible. A considerable improvement would be obtained if only four engines were used, but nevertheless three outstanding difficulties would remain. These would be the difficulty of incorporating reverse thrust, the access problem-implying a need for different upper and lower powerplant arrangements, and the access panel cutouts in the wing box. It is reasonable to conclude that a great deal more work is necessary to enable such an installation to be designed satisfactorily.

5.3 Conclusions

1. The integrated layout is the logical one for a large slender aircraft design, the main problems being the provision of acceptable windows and rear entry. These should not be insurmountable.

2. The benefits of the double wall cabin arrangement are only marginal, but if it is used the "treble bubble" cross section makes excellent use of available volume.

3. A buried rear engine installation is fraught with difficulties, although some of these are eliminated when four instead of six engines are installed.

PART 2

MACH 3.0 AIRLINER, PROJECT A-62

6.0 Choice of Configuration for the A-62

The specification of the Mach 3.0, A-62 design was fixed by the decision to investigate an aircraft which was directly comparable to the earlier A-60 project in all aspects except cruising speed. Experience with the slender wing configuration of the A-60 had shown that it became exceedingly difficult, if not impossible, to reconcile low speed performance requirements with a useful subsonic leading edge at cruise Mach numbers in excess of about 2.4. One particular difficulty was found to be the elevator power necessary to raise the nose during take off, but the low useful lift coefficient is also of considerable significance. The application of variable sweepback to the A-62 was ruled out from the outset for two reasons. Firstly it was considered that insufficient background information was available, and secondly the incorporation of variable sweepback would have rendered invalid a direct comparison with the A-60 project. It was therefore necessary to accept a supersonic leading edge during cruise, and various configurations were investigated on this basis.

Although a tailless design was attractive in that it enabled a minimum volume to be attained in conjunction with a favourable area distribution over a wide speed range, the relatively high trim drag associated with either the subsonic or supersonic cruise largely offset these advantages. A conventional layout with a rear horizontal stabiliser suffered from severe area ruling difficulties without really eliminating the trim drag problem. The use of a canard configuration was promising in that it enabled the trim force to be usefully imployed in contributing to the lift of the aircraft, thereby reducing the trim drag, without greatly complicating the area ruling problems as the foreplane is not located in a critical region. Against this it was anticipated that dynamic stability problems would be encountered, but evidence from various sources indicated that these were not likely to be prohibitive providing the foreplane span and area were kept small relative to those of the wing. A further advantage of the canard arrangement is the good longitudinal manoeuvre response. Having regard to all aspects of the problem the canard arrangement was selected as that which appeared to give the best compromise.

The wing planform was chosen to be a 50° delta since this was found to give a good compromise between the lowspeed, transonic, and cruise performance requirements. It was thought that both the foreplane and the control surface attached to it should have reasonably high lift curve slopes and therefore a basically unswept planform having an aspect ratio of three was chosen. In fact it was found subsequently that this was not as critical as had been anticipated and the area rule shaping of the body could have been simplified by introducing sweepback to the foreplane. The influence of the foreplane upon the area distribution was very important at transonic speeds. The location of the powerplant presented some difficulties in the layout of the aircraft. The buried engine installation of the A-60 design proved to be unsatisfactory and it was therefore decided to use podded powerplants for the A-62. Initially six engines, each of 20,000 lb. sea level static thrust, were mounted in individual underwing pods, but this arrangement imposed serious restrictions upon the location of the wing trailing edge control surfaces. The final solution used only four engines in conjunction with inboard ailerons on the wing, and elevators on the foreplane.

The aerodynamic performance of the aircraft was analysed in detail by Burrow [14]. This included an optimisation of the area distribution. The overall stability was also investigated[15] and, as expected, some regions of instability were discovered. These occur mainly in the transonic region and are well within the capability of an autostabilisation system, but cruise directional stability was found to be marginal. This latter difficulty was rectified by incorporating moving wing tips in the design. These fold down during the transonic phase, in a way which helps to smooth out the aerodynamic centre movement, and add a vital increment to n_V at Mach 3.0.

6.1 The A-62 Design

A general arrangement of the A-62 project is shown in Figure 18. The 50° delta wing has a basic area of 4,500 sq. ft. for the cruise configuration. In this form the wing tips are drooped, and when they are raised for low speed flight the area is increased to 4,650 sq. ft. and the span is changed from 102 ft. to 118 ft. A slight sweep forward of the wing trailing edge is incorporated in the planform to enable the aileron hinge line to be perpendicular to the aircraft centreline. The maximum take off weight was estimated to be 390,350 lb. when the aircraft was loaded to carry 108 passengers over 3,250 n. miles range. The maximum corresponding landing weight is 232,000 lb. At these weights the estimated take off and approach speeds are 200 knots and 145 knots, the latter being somewhat less than that anticipated for the A-60 project. Nearly 10,000 ft. of runway are required for the take off and about 4,400 ft. are necessary to land from 50 ft. height without the use of reverse thrust. The wing section chosen was basically a 4.5% thick beconvex shape based on 110% of the actual chord, which gives a 4.95% thick aerofoil with a blunt trailing edge. A small nose radius was added to improve low speed characteristics. The cruise is thrust limited, the lift coefficient of 0.12 giving a lift-drag ratio of 7.20. The maximum lift-drag ratio of 7.33 should be capable of some improvement with further refinement of the shape.

Each of the four underwing podded powerplants develops 30,000 lb. sea level static thrust. The engines are of the bypass type with cold stream burning to 1150°K for the climb and supersonic cruise flight phases. An axisymmetric translating centrebody intake forms the nose section of each pod and reverse thrust capability is incorporated with the convergent-divergent nozzle. Engine driven accessories are mounted in the wing, immediately above the engine location.

The cabin is tapered to a maximum diameter at the aft end, as is shown in Figure 19. The taper was determined by the area distribution, and it effectively causes the cabin to be divided into three sections having four, five and six abreast seating. The layout shows 120 seats at 33 inches pitch with toilets located at the front end and between the cabin sections. Passenger entry doors are positioned at these latter two points. Freight is carried in a hold which is located below the cabin floor. The crew cabin is separated from the passenger cabin by a gangway which passes below the foreplane and can be sealed at both ends by pressure doors should an emergency occur. Apart from the nose and rear extremities the fuselage is of circular cross section.

The main undercarriage is attached near to the leading edge of the wing and it retracts sideways so that the wheels are stowed in the fuselage just aft of the cabin. The nosewheel is stowed below the cabin floor. A large proportion of the fuel is carried either in the wing or in tanks located in the fuselage behind the mainwheel bay, but in order to balance the aircraft it was found to be necessary to use the volume of the foreplane and also that of the fuselage below it. An extended root leading edge to the wing to give the correct area distribution would have eliminated the need for this undesirable forward fuselage tank and the tapered cabin.

7.0 Detail Specification of the A-62

The detailed specification of the A-62 project is contained in Appendix D. The predicted weight breakdown, together with some values estimated as a result of the work of the students are given in Table 1.

8.0 Description of the Structure of the A-62

The basic requirements for the design of the structure of the A-62 were similar to those used for the A-60. The specified life of 30,000 hours includes 20,000 hours in the cruise condition when much of the structure is subjected to a temperature of the order of 280°C. An average flight duration of one hour was assumed. Both titanium and stainless steel were considered for use as the main structural material. A preliminary investigation showed that the use of high grade titanium alloys enables a lighter structure to be designed than if stainless steel is used. However titanium alloys were ruled out on the basis of material cost, the fabrication difficulties associated with high grade alloys, and general lack of experience in the use of the material. Stainless steel, grade FV520, was chosen as a more straightforward material having favourable fatigue characteristics at high temperature, and the structural design was based upon it in spite of the implied weight penalty. Sheet metal components are fabricated in FV520S, the austentitic form, and forged items in FV520B, the martensitic form. The main departures from this occur in the undercarriage which largely uses S99, the windscreen structure which is cast in EN55, and the limited use of light alloy, L73, sheet for some internal components.

An analysis of the fatigue and creep of the structure, with particular respect to the cabin region revealed that creep was not of great significance[16]. The maximum normal flight tension stress in FV520 is 32600 p.s.i. The most critical aspect of the structural design was found to be the methods of fastening and joining, and in the majority of the structure spot welds are used. The mechanical design problems were greatly aggravated by the lack of available information on bearing performance, this being especially critical for the flying control systems. The detail design was based upon the use of S99B chrome plated journals running in phosphated S99B bearings which have been shown to be capable of operation at 20,000 p.s.i. bearing pressure at 300°C. However in some cases a bearing pressure of 35,000 p.s.i. was found to be desirable in order that the components could be reduced to reasonable proportions, and it was thus necessary to consider some form of cooling.

The disposition of the main structural members is shown in Figure 20. The aircraft is structurally conventional and no unusual loading cases were encountered. The aeroelastic characteristics were investigated [17] [18] with the aid of a dynamic model[19], which has a built up light alloy wing with solid wooden body, fin and foreplane. A special theoretical analysis of the landing dynamic loads was also undertaken.[20]

8.1 Fuselage

The maximum unfactored longitudinal fuselage shear force of 190,000 lb. occurs at a section coincident with the wing root leading edge, and is due to the effect of a gust during subsonic diversion flight at 209,000 lb. weight. The maximum unfactored bending moment occurs at the same section but is caused by a gust during the initial climb at 378,000 lb. weight. It has a magnitude of 7.8×10^6 lb. ft.

The majority of the outer surface of the fuselage is designed to use a corrugated reinforced skin construction. This was found to be the best way of overcoming the problem of stabilising the relatively thin FV520 outer skin. In most places the internal corrugations have the same thickness as the skin to which they are attached by spot welds. Seam welds are used for the skin joints. At the forward end of the fuselage, in the region of the crew compartment, the corrugated panels only occur at the upper and lower portions of the fuselage cross section. Top hat stringers having a mean pitch of 4.0 inches are used on the sides, their thickness varying from 29G to 24G. The skin thickness varies from 30G at the nose to 25G at the aft end of the crew cabin and the 0.4 inches deep corrugations having a pitch of 1.50 inches between the rows of welds. The skins are supported by pressed frames located at 15 inches pitch, each alternate one providing a support for the floating crew floor. The latter uses 0.25 inch deep end grain balsa which has 26G and 30G upper and lower L73 facing sheets. Special top hat stringers, each having a cross sectional area of 0.29 sq. in. are located across the top and bottom of the cutout which is necessary to accommodate the plug type crew door. The door itself is of double wall construction with pressed internal stiffeners. The opening mechanism causes it to move inwards and then rotate before it passes out through the cutout. A 30G domed front pressure bulkhead is welded to a machined ring to enable it to be attached to the skins. The windscreen structure consists of I section EN55 steel castings which are attached to forged FV520 frame members. Cast steel frames support the transparencies and provide an alternative load path should a failure of a main pillar occur. An air gap of 0.4 inches separates the 0.75 thick fused silica outer panel from the internal panel of toughened glass-vynal laminated construction. This 0.75 inches thick internal panel reacts cabin pressure loads and gives protection in the event of a bird strike.

In the region of the foreplane the fuselage acts as an integral fuel tank. The 26G corrugated skins are supported both by frames placed at 20 inches pitch and intercostals. Corrugated FV520 construction is also used for the inner tank walls which form a gangway between the crew and passenger cabin. The 30G tank roof is flat but fluted laterally to relieve thermal stresses. The tank end closures are domed in shape and 26G thick. They serve also as pressure bulkheads for the two cabins, the gangway being closed in emergency by pressure tight doors. The gangway floor is separated from a flat tank sealing skin by a space which is cooled by toilet discharge air. It is connected to the fuselage rigidly at the front end only. In all fourteen points are used to attach the foreplane above the tank roof. Of these there are twelve swinging lift links, six on each side, which coincide with the fuselage frames. The links are forged in FV520 and have ball end joints. Drag loads are reacted by a multi-jawed swinging link located on the aircraft centreline at the foreplane front spar. This link shares the side loads with the other attachment which is located at the foreplane rear spar centreline. It consists of a forged tongue which is free to slide both longitudinally and vertically.

Over the main cabin region the frames are located at 12 inches pitch and the skin thickness varies from 25G at the front to 22G at the rear. The longitudinal skin joints use a Tee shaped FV520 extrusion as a butt strap with spot welds placed at 0.5 inches pitch. A flat rear pressure bulkhead of corrugated sandwich construction closes the rear end of the cabin. This is 1.0 inch deep and has 26G thick facings and core. The design of the floating floor is illustrated in Figure 21. It is constructed from a series of panels which are located only at one end and are connected by sliding joints. Some 75% of the floor length is unpressurised and in this region it is manufactured from end grain balsa with 24G thick L73 facings. Over the nosewheel bay the floor reacts pressure loads and here it is of 0.6 inches deep brazed steel honeycomb construction with 26G facings. The passenger entry doors also use steel honeycomb for their outer surfaces, but in this case it is 0.25 inches thick with 24G and 20G inner and outer facings respectively. These plug type doors are located above the centreline of the section and open by lifting and then swinging outwards. Window design in a Mach 3 aircraft is difficult and it has been estimated that their incorporation introduces a weight penalty of over 2,000 lb. The suggested arrangement has separate glasses to resist heat and pressure loads, with an inner protection lens behind which cooling air is passed. The nose undercarriage is attached to a stiffened, sandwich bulkhead by deep forgings. The layout of this member is shown in Figure 22 together with the subsidiary bulkhead at the other end of the nosewheel bay.

Aft of the cabin the fuselage decreases in size and merges with the wing and the fin. This region houses both the wheels and the rear fuselage integral fuel tanks. The 24G skins are supported by frames placed at 12 inches pitch. The undercarriage bay is divided by a longitudinal sandwich web which has 24G faceplates and 16G edge members. At its lower edge this web is attached to a keel member which is built up from 10G angles with 10G cover plates. The outer edges of the undercarriage bay are reinforced by tapered 14G angle section longerons. The fuel tank is closed at its end and divided longitudinally by flat, 20G corrugated bulkheads. The main, rear, wing box passes below the extremity of the fuselage, but the spars forward of the undercarriage bay are effectively extensions of the fuselage frames. The fin terminates at five bulkheads which are mounted on the wing. These are 14G thick with 16G reinforcing corrugations. Frames are located between them at 10 inches pitch.

8.2 Wing

The wing structure consists basically of 20 spars, 15 of which are aft of the undercarriage bay, and 16 ribs on each side of the aircraft. The innermost rib slopes in plan and coincides with the side of the fuselage, whilst the two rearmost spars occur only outboard of the ailerons. The maximum unfactored wing spanwise bending moment of 7×10^6 lb. ft. occurs when a gust is encountered during the initial subsonic climb, but it is only slightly more severe than the 2.5g manoevre case.

The main wing structure, which is the portion aft of the undercarriage bay, is designed as a single component with the box passing below the rear fuselage structure. It is attached at both skin and frames. At the rear of the aircraft the fuselage merges into the fin which is also attached at appropriate rib and spar positions. A large part of the wing is used as an integral fuel tank and it was found to be necessary to provide some form of insulation. For this reason steel honeycomb sandwich was chosen for the wing skins rather than corrugations. All the honeycomb panels use a

0.5 inches core of 0.8 inches depth with equal facings. This facing thickness varies from 20G at the centre of the root chord to 32G along the leading edge spar and 25G along the trailing edge. Two schemes were considered for the arrangement of the individual skin panels. One used spanwise units of up to 44 ft. length and 5.5 ft. width, and the other chordwise units which have a maximum size of 16.5 ft. by 6 ft. Although the former is structurally preferable, the latter represents a more feasible proposition from the production point of view. Skin joints are made by brazing the panel edges together in situ. Spar and rib booms are contained within the sandwich skin panels and are used to attach them to the corrugated webs. Where possible the spanwise booms are continuous and the chordwise rib booms are intercostal with local thickening at the joints. Spar web thickness varies in the range of 20G to 24G. Reinforced plate ribs are provided at tank ends, the engine pod pick up points, and the control surface attachments. In the latter case the skins are locally reinforced to assist in the diffusion of the rib end loads.

The main undercarriage pivot and sidestay are attached to the two spars on either side of the undercarriage bay. Wing structure forward of this area is similar in construction to that used for the main wing box, but the leading edge and two front spars terminate at the cabin side where they are pin jointed to appropriate frames. At it's outboard end the wing terminates in a rib which forms a base for the moving tip hinges. The hinges themselves are located at the extremities of three of the six spars which terminate at this rib, with the actuation jacks arranged in pairs on either side of each hinge. The spar end fittings, which incorporate the hinges, locking pins, and actuator attachments, are FV520 forgings. The wing tip structure includes 5 spars and 5 ribs, the skin panels thus formed being approximately 2.7 ft. square. These skins have corrugated reinforcement and are 24G thick. All the webs are corrugated and the leading edge is of full depth brazed honeycomb construction with 32G skins. The wing tip is locked in the down position mechanically, but relies upon hydraulic pressure for the uplock. It has been estimated that the total weight penalty arising from the incorporation of the moving tips is 1,780 lb. of which roughly 65% is in the fixed part of the wing.

Ailerons

The maximum aileron load of 45,200 lb. per side arises in a rolling pullout at Mach 3. The construction of the ailerons and the other control surfaces is based upon the use of North American "Spacemetal". This is a flat stainless steel corrugated sandwich material of 0.15 inches depth with a 47G (0.002 inches) thick core welded to 38G (0.006 inches) thick facings. In the present application the corrugations are arranged to run in the chordwise direction and the joints are made by crushing the core locally or inserting doubler plates, and welding. The two aileron spars use Tee shaped extruded FV520 booms with 24G tension field webs. Spaced at 12 inches pitch the ribs are welded, braced, frameworks in 20G, T45 tubes of 0.75 inches diameter. Three forged hinge fittings are used, the hinges themselves being of the plain variety mounted in a spherical cage so that a second surface is available should primary seizure occur. The hinge line is located well below the section depth and the actuators are positioned just inside the top skin.

8.3 Fin and Rudder

The maximum fin and rudder loads arise when the rudder is operated at the damped natural yawing frequency of the aircraft. A maximum unfactored load of 215,000 lb. occurs on the whole assembly at M = 0.9 and the design diving speed whilst the rudder alone is subjected to 28,200 lb. at M = 1.41.

The fin construction is based upon five swept spars which coincide at the root with the appropriate wing spars and fuselage bulkheads. At approximately one chord out along the span the two intermediate spars are terminated at a rib. The problem associated with the mounting of the spars on the flexible wing structure was investigated separately [21] [22]. Each spar has a plate web which varies in thickness from 13G at the root to 15G at mid span, and machined, tapered, booms which are "puddle" welded to the skins. At the root the booms possess a substantial cross sectional area and are used to collect the skin end loads and transmit them to the wing spars. The skins are reinforced by 16G Zed section stringers which have a pitch of 2.3 inches outboard and 1.5 inches at the root. These are spot welded to the skins whose thickness varies from a maximum of 12G at the root to 18G at mid span. The end load carrying structure is supported by chordwise plate ribs at approximately 18 inches pitch which are provided with castellated skin attachments between the stringers. Closely spaced riblets placed normal to the leading edge spar maintain the nose shape of the aerofoil section.

Rudder

The rudder is divided into three spanwise sections and each is provided with two hinges. The lower one of these in each case is a skew hinge through which the rudder is operated. The rudder skins use "Spacemetal" which is supported by ribs placed at 21 inches pitch, normal to the hinge line.

8.4 Foreplane and Elevators

The maximum normal acceleration and subsonic gust cases give similar foreplane loading cases. The unfactored load is approximately 140,000 lb. and the corresponding root bending moment is 5×10^6 lb. ft. Like the wing the foreplane is used as an integral fuel tank, and hence it was found to be desirable to use brazed steel honeycomb skin panels. The basic structure employs six spars with six ribs on either side of the centreline, the box being located between 18.5% and 65% of the chord. The spanwise skin panels are tapered and have maximum dimensions of 16.5 ft. by 1.5 ft. A constant core depth of 0.7 inches was chosen with equal facings which vary from 18G thickness at the root to 34G at the tip. The mean rib pitch is 40 inches. Plate webs and Tee section booms are used for the two outer spars, but all the other webs are corrugated and intercostal between intersections for simplicity of production. The elevator is attached at all five of the outboard ribs and a detachable panel is provided in the top surface behind the rear spar for access to the hinges and actuators. The twelve fuselage attachment lift links are connected to brackets mounted off the lower surfaces of the spar booms.

Elevator

The construction of the elevator is very similar to that of the ailerons in spite of the fact that it is relatively shallow. The "Spacemetal" skin panels are used in conjunction with flanged 22G plate ribs placed at a mean pitch of 11 inches, and two 24G tension field spars. As on the aileron the hinge line is located near to the lower surface of the section. The three sets of actuators are placed on the central three of the five hinges.

8.5 Undercarriage

The main undercarriage units are attached to the wing near to its leading edge and retract sideways so that the eight tyred bogie assemblies are stowed in the fuselage immediately behind the cabin. The highest vertical and side loads arise during ground manoeuvring, when the unfactored values are 269,000 lb. and 134,500 lb. respectively. Drag loads reach a maximum of 118,500 lb. per unit during a high drag landing. The layout of the main undercarriage is shown in Figure 24.

The main shock absorber is a liquid spring which uses a silicone base oil. It is placed inside the telescopic main leg which is an S99 forging. The top of the leg is Y shaped and ends in a wide trunnion which is located between the faces of the adjacent spar webs. Both the trunnion and the rear member of the Y piece are manufactured separately from the main leg fitting. A hop damper is placed between the bottom of the leg and the tubular S99 bogie beam. This unit assists in dissipating the vertical landing energy and also serves to trim the bogie so that the rear wheels touch the ground first. The main leg is supported by an articulated Y side stay which is cross braced back to the leg from its' joints.

Each of the titanium wheels carries two tyres and is mounted on an axle assembly which is fabricated separately from the bogie beam. Disc brakes are incorporated in each of the four wheels on a bogie, copper being preferred as the disc material. The downlock is located at the side stay joint and the uplock is on the bogie beam, both being operated by auxiliary jacks. The wheel bay is closed by two sideways opening doors which are hinged on the keel member. A similar arrangement is used to seal the wing portion of the undercarriage bay, the doors being hinged on the bay end rib. The bay is cooled by gaseous nitrogen drawn from the radio and underfloor compartments, the maximum temperature being 160°C.

Nose Undercarriage

A general arrangement of the nose undercarriage appears in Figure 25. The maximum unfactored vertical load of 104,500 lb. occurs during dynamic braking whilst the normal three point landing gives a side load of 17,450 lb. and a high drag landing results in a drag load of 41,800 lb. The telescopic unit has a separate liquid spring shock absorber which is located within the sliding members. The four tube arrangement is completed by the addition of a torque tube to transmit steering moments. The steering mechanism is located as far down the leg as is allowed by the geometry. It is illustrated in Figure 26. The steering torque loads are transmitted across the sliding member by torque links, and good castoring performance is ensured by 9 inches of positive trail. Side loads are transmitted to the fuselage by the Y shaped upper casting, which like the majority of the rest of the components, is fabricated in S99. An articulated radius rod acts as a drag strut. The leg is locked down by an internal lock in the retraction jack which is attached at the joint on the radius rod. Retraction involves a backwards and upwards motion of the leg, the wheels being rotated through 90° by the steering mechanism. When retracted the leg is stowed across the bay at an angle of 3.25° so that the wheels lie on the aircraft centreline and the best use is made of the available depth. The unit is locked up by an auxiliary jack operated pin which engages with a lug placed on the rear face of the leg. The bay is closed by a pair of doors which are hinged along the edge members, and it is cooled by discharge air.

9.0 Project A-62 Installations and Systems

The locations of the main items used in the systems and installations of the A-62 design are shown in Figure 19.

9.1 Power Plants

The four bypass powerplants are mounted in separate pods below the wing and include axisymmetric variable intakes and convergent-divergent nozzles. Each assembly is suspended from the wing on a pylon which structurally consists of 9 swept spar members and fairing skins. The spars have 16G thick FV520 webs and angle booms, and attach directly to double wing ribs. At their lower ends they terminate in frames which form the basis of the engine mounting and cowling structure. This is of skin-stringer construction with substantial longitudinal forgings at the main engine mounting points. These are a pair of trunnions, one of which is arranged to transmit side loads and the other is free to slide laterally. The trunnions are mounted into bearings located on cams so that rotation enables adjustment to be achieved. The front engine pick up is a swinging link which attaches directly on to one of the pylon spars.

A substantial D.T.D. 705 casting is attached directly on to the front of the outer engine casing and this provides support both for the outer intake cowling and the centrebody. The former is of conventional construction with 22G thick double skins, channel section rings, and eight top hat stringers. At the extreme nose it terminates in an annular forged lip. Additional 18G stiffeners are placed in areas where shock waves impinge upon the inner wall. Six auxiliary doors located round the circumference of the intake are used to supplement and control the airflow. The centrebody is supported by seven radial vanes on the engine front face casting. This also uses a skin-stringer construction. The inner, fixed tube is a circular cylinder with 20G skins and 22G top hat section stringers. On the outside of its surface there are six guide rails for the outer, moving section. Apart from a solid machined nose portion the moving section is similar in construction to the inner tube. Four rollers engage with each of the guide rails, and the intake is operated by a pneumatic jack which is supplied by engine bleed air. The stroke of the jack is 16.5 inches and its large diameter of 11.0 inches is necessary because of the low supply pressure in some flight conditions. Boundary layer air is bled from the centrebody and passed through the support vanes before ejection. Apart from the castings, FV520 is used throughout the intake.

The nozzle assembly is fabricated in Nimonic 95. It is attached to the aft face of the engine cowling at four points and incorporates a moving centrebody and outer ejector tube as well as reheat. Provision is made for reverse thrust.

9.2 Auxiliary Power Supplies

The auxiliary power supplies for the A-62 were the subject of a special study[23]. The proposed solution uses both engine driven equipment and an airborne auxiliary power unit. The latter is mainly provided for engine starting and emergency use. The normal system consists of two hydraulic pumps and one 100 KVA alternator driven by each engine, and located in the wing. A simple mechanical drive is used without a constant speed unit.

The hydraulic pumps operate at 4,000 p.s.i. and are arranged to supply three separate systems each of which is capable of providing one half of the maximum power requirement. Each pump has an output equivalent to 100 H.P. and it is intended that "Silcodyne M" fluid should be used to enable the system to operate at up to 300°C. Accumulators are provided to cover the period between a supply failure and the A.P.U. being brought to full power. This has two pumps which are sufficient to meet the essential flight demands. The undercarriage circuit has a separate emergency accumulator. Seals present a problem at 300°C, but it is anticipated that the use of the continuous metal ring type for sliding seals and neoprene for static seals would prove to be satisfactory.

The four shaft driven alternators feed six busbar systems. Two of the alternators feed constant frequency A.C. inverters and these two, with the others, feed four separate inverters which give both variable frequency A.C. and D.C. supplies. It is intended that solid state units should be used. The A.P.U. is equipped with one 40 KVA alternator for emergency use. A battery is provided for starting the A.P.U. and as an emergency D.C. source.

The air intakes are operated pneumatically by air bled directly from the engine compressor.

9.3 Flying Controls

The three sets of flying control surfaces and the moving wing tips are all fully power operated from the hydraulic system. A mechanical signalling system was chosen because of its inherent reliability in spite of the difficulties associated with the expansion of the airframe. These difficulties were overcome either by locating the stainless steel control runs where they would expand with the structure, or preferably by using torsion controls which are not normally sensitive to temperature effects. Cables are used for the fore and aft runs in the fuselage and tension regulators are incorporated in the circuits. It has been estimated that the run from the cockpit to the foreplane necessitates 2.8 inches of compensation and that to the ailerons and rudder a total of 11.2 inches compensation on two regulators. All the longer spanwise runs are of the torque tube variety.

The control surfaces are operated directly by the hydraulic jacks which are arranged in groups of three to coincide with the three hydraulic systems. Like these systems, each jack can provide half of the power required by the group of three. Each aileron is operated by three jack having an individual capacity of approximately 100,000 lb. The control valves are in line with one another and are connected by a torque tube. Twist is not significant due to the small valve operating loads. In the case of the elevators the arrangement is similar with three jacks on each side, which produce 35,000 lb. of thrust individually. The rudder system is somewhat different in that a single jack is used to operate each of the three rudder sections through the skew hinge assemblies. Thus in the event of a rudder jack failure only two parts of the rudder can be controlled, whereas in the case of the aileron or elevator the whole surface can still be moved to its full extent.

Bearing design, both for control hinges and jack ends, presented a considerable problem. Satisfactory compact designs were only achieved by assuming that working stresses of the order of 35,000 p.s.i. could be achieved in cruising flight. Nitrogen cooled bearings were considered as a means of achieving this. The bearing problem is especially severe at the wing tips because of the small section depth. It was found to be necessary to use three pairs of jacks, each jack being capable of 66,000 lb. of thrust.

9.4 Fuel System

The fuel system has a total capacity of approximately 23,700 imperial gallons. Of the eleven integral tanks, two are in the foreplane, one in the fuselage immediately below the foreplane, two in the rear fuselage and the rest are in the wing. A booster pump feed system is used and it is designed so that a single failure can occur without any loss of available fuel. This is achieved by pump duplication and cross feeds. The side wall mounted, hydraulically driven, pumps cater for the maximum delivery of 30,500 gallons per hour which is demanded at the start of the climb. The centre of gravity of the aircraft is controlled by proportioning the fuel supply from the various tanks, the pump motor inputs being varied as required for this purpose.

A study of the problem of fuel temperatures indicated that it should be possible to develop a fuel capable of being heated to 140°C or 150°C without the onset of serious detrimental effects. Both theoretical [24] and experimental [25] investigations were undertaken to specify the tank characteristics necessary to achieve these maximum temperatures. All the tanks are pressurized by a 100% nitrogen system. It is anticipated that if nitrogen is also injected into the fuel during refuelling, the oxygen content of the fuel tank gases should fall below 2.5% and there would be no danger of explosion. The tank vents lead into a gallery which runs along the top of the fuselage and is provided with surge tanks and relief valves at either end. The tank pressure is controlled to be 2 p.s.i. above ambient or 6 p.s.i. absolute, whichever is greater. Very little tank insulation is necessary in the wing and foreplane due to the use of steel honeycomb sandwich skins. It is suggested that a fluorcarbon plastic would be suitable for use in the fuselage tanks, but satisfactory bonding would have to be developed.

Fuel flowmeters are included in the system to supplement the capacitance type fuel gauges. It takes twenty minutes to fill the tanks from the normal reserve level when 500 gallon per minute bowsers are connected to each of the two undercarriage bay refuelling points. The fuel can be jettisoned by the booster pump system at the rate of 1.28% of the aircraft all up weight per minute, so that the maximum landing weight can be achieved 26 minutes after take off. The jettison pipe is located at the extreme rear of the fuselage.

9.5 Environmental Control System

The design of the environmental control system for a passenger aircraft flying at Mach 3 and altitudes above 70,000 ft. presents a severe problem. The solution suggested for the A-62 design is unusual in that it is entirely self-contained and does not rely at all upon air taken in from outside. The system is a cryogenic one with liquid oxygen and nitrogen stored in separate containers under the rear cabin floor. The latent heat of evaporation of these liquids is used to cool the aircraft, the distribution system being shown in Figure 23.

Cold air is obtained by mixing the evaporating nitrogen and oxygen and this is heated in a muffler-mixing device as it is combined with recirculated cabin air. The mixed gases are fed into the cabin at the roof racks at the rate of 1.2 lb. per minute for each passenger, and they are extracted at floor level. Before recirculation the water vapour content is adjusted, the carbon dioxide removed by passage over lithium hydroxide, the impurities filtered out and the oxygen content corrected. Air from the galley and toilet areas is not recirculated, but is used to cool the nosewheel bay before it is dumped overboard. Gaseous nitrogen is used to cool the freight bay and electronic equipment, and this is then discharged through the mainwheel bay. The system for the cabin is duplicated and the pilot has a separate supply for his pressure suit. Emergency oxygen is supplied at all passenger and crew positions. In order to reduce the insulation required for the cryogenic containers the oxygen tank is located within the nitrogen one.

The cabin wall is insulated and cooled to reduce the rate of heat transfer. An air gap is left between the outer skin and a layer of 'Thermoflex' insulating material which has a foil radiation facing on its outer surface. Cooling air is passed between the thermoflex and the inner cabin lining, the whole assembly having a depth of 8 inches. The cabin heat load at the end of the Mach 3 cruise has been estimated to be 133,000 C.H.U. per hour. It has been calculated that the loss of a window at 76,000 ft. altitude would cause decompression to occur in 52 seconds, whilst if an escape hatch or door failed the times would be 20 seconds and 3 seconds respectively. The last two figures are not tolerable and fail safe measures are essential in the design of these components.

The system chosen proved to be heavy largely because of the 7,000 lb. of cryogenic liquids. However against this must be set the advantages of the system, which are primarily the elimination of any possibility of ozone poisoning, and no engine losses or additional cooling requirements. It is likely that a mixed system, using either engine bleed air or engine driven compressors for subsonic flight and cryogenic liquids for supersonic flight, would prove to be a favourable compromise.

10.0 Special Aspects of the A-62 Design

10.1 Cabin Layout and Fuel Tanks

A tapered cabin was used in the layout so that a favourable area distribution could be achieved, but it has the serious disadvantage of restricting the variation of seating arrangement. Although ample total volume is available for the required fuel much of it is aft of the centre of gravity and a satisfactory distribution was only achieved by introducing an undesirable front fuselage tank. A modified layout with a basically parallel cabin and extended leading edges to the wing would have been a better arrangement. Not only would this have improved the flexibility of seating layout, but it would also have added additional wing tank volume, forward of the centre of gravity. It might also have enabled the root wing section depth to be increased sufficiently to house the main wheels with a consequent improvement in the fuselage structure.

10.2 Materials Problems

As was anticipated from the outset the major materials problems were found to be the non-metallics and bearings. Investigations showed that whilst in many cases suitable non-metallics were not immediately available, existing developments could lead to a satisfactory solution in the not too distant future. The bearing problem is one of degree and current technology is such that it is not possible to achieve satisfactorily compact and light designs. New developments are therefore required.

10.3 Constructional Methods

The corrugated reinforced skins used throughout the fuselage proved to be a very satisfactory solution to the problem of light gauge steel design. On the other hand the wing sandwich panel construction must be viewed with considerable reservation. This construction was chosen for reasons of fuel tank insulation but it suffers from considerable internal thermal stress and it is difficult and costly to produce. It is felt that corrugated reinforced wing skins with a separate internal insulation would have given a simpler and lighter structure. The use of 'Spacemetal' skin panels enabled relatively light control surfaces to be designed.

10.4 Moving Wing Tips

The moving wing tips proved to be valuable in improving both transonic longitudinal stability and cruise directional stability. The major design problem was the small local depth of the wing section which was available for hinges and actuators. The total estimated weight penalty of 1,780 lb., which is less than 0.5% of the all up weight, dows not seem to be excessive.

10.5 Fuel System

It is not possible to use the fuel system of an aircraft cruising at Mach 3 as a heat sink, and indeed it is necessary to minimise the rise in fuel temperature as much as possible. Considerable tank pressurization is required and the pure nitrogen system appears to be the most promising way of achieving this. Although a booster pump feed system was proposed for the design it is felt that the basic tank pressurisation level is sufficiently high to warrant consideration of a pressure feed system.

10.6 Environmental Control System

The choice of a cryogenic environmental control system was made after due consideration of the total aircraft heat load and the possibility of ozone poisoning. It is a very attractive proposition but as designed is somewhat heavy. A mixed system using more conventional techniques for the subsonic flight phases is likely to prove to be a satisfactory compromise.

10.7 Conclusions

1. The overall layout of the aircraft would have been improved by using extended wing root leading edges in conjunction with a parallel cabin.

2. Although suitable non-metallics are not immediately available for a Mach 3 cruise aircraft, they should become ready in the relatively near future. Developments in bearing materials and techniques are required.

3. Corrugated reinforced skins are a satisfactory solution to the problem of light gauge design, but sandwich construction is not without difficulties.

4. The moving wing tips were a valuable addition to the aircraft and the weight penalty introduced by them is reasonable.

5. Although a booster pump feed fuel system was proposed, a pressure feed system is worthy of consideration.

6. The cryogenic, sealed, environmental control system is attractive but should preferably be used in conjunction with a conventional system for subsonic flight.

PART 3

11.0 Comparison of M = 2.2 and M = 3.0 Supersonic Airliners with Special Reference to the A-60 and A-62 Designs.

A comprehensive comparison of the characteristics of supersonic airliners designed to cruise at M = 2.2 and M = 3.0 has been made by Porter[26]. This comparison was applied particularly to the A-60 and A-62 project design studies described in Parts 1 and 2 of this report. For convenience the investigation dealt separately with the various aspects of the problem.

11.1 Aerodynamic Considerations

The overall aerodynamic aspects of supersonic airliner design are considered in Part 1, Section 1 and Part 2, Section 6 of this report. The most important conclusion from these discussions is that, apart from the possible use of variable sweepback, it is not likely to be possible to have subsonic leading edge flow conditions for cruise speeds in excess of M = 2.4. This implies that aircraft operating at the higher cruise Mach numbers will have a lower aerodynamic cruise efficiency than would otherwise be the case, but the lowspeed characteristics may well be preferable, in spite of a higher basic aircraft weight (see Table 1).

11.2 Propulsion Considerations

The thrust requirements arise from consideration of three distinct flight phases; take off, transonic acceleration which must occur at as high an altitude as possible, and cruise. Aerodrome noise level must be minimised, although the increased thrust to weight ratio of all supersonic airliners relative to subsonic ones implies a more rapid climb and hence a less serious noise problem outside the aerodrome boundary. It would appear that a M = 2.2 cruise design is best based on the use of turbojet engines or turbofans with a low bypass ratio, low reheat being used in either case during transonic acceleration. On the other hand a relatively higher bypass ratio turbofan with moderate reheat for climb, transonic acceleration and supersonic cruise is more suited to the requirements of a Mach 3 design. The latter is basically a quieter engine at take off and has a low specific fuel consumption for subsonic cruise when reheat is not used. The former advantage is likely to be offset by the higher thrust required for the relatively heavy Mach 3 aircraft.

Variable, multi-shock intakes are necessary on both designs, but are more critical and complex at the higher cruising speeds. Both types of supersonic airliner must be fitted with variable geometry exhaust nozzles, but again the Mach 3.0 design is more critical.

11.3 Materials and Structures

One of the major advantages of the Mach 2.2 supersonic airliner is that it is possible to use a basically light alloy airframe with conventional, relatively simple constructional techniques. Flight at higher Mach numbers for extended times implies the use of either titanium or stainless steel, the former being preferable from the weight aspect. Both of these materials introduce constructional and manufacturing difficulties which, whilst they are by no means insurmountable, must inevitably increase the initial and maintenance costs.

Non-metallic and bearing materials become an increasing problem as the cruise Mach number increases. It is probably true to say that because of this the satisfactory operation of airliners at cruise speeds of the order of Mach 3 will not be possible for some appreciable time and that during this period Mach 2.6 or 2.7 is likely to be an upper limit.

The integrated layout, slender delta configuration proposed for the M = 2.2 cruise design enables a high structural efficiency to be achieved, and this should not be significantly less for a design using a more conventional fuselage in conjunction with a slender wing. A canard design, as proposed for the Mach 3.0 design, is less attractive in this respect.

11.4 The Sonic Boom

The sonic boom question is one which will be a subject for discussion for many years to come, and it is virtually impossible to reach any absolute conclusions. It is, however, possible to compare aircraft on the basis of similar assumptions. The two critical considerations are the ground pressures caused during transonic acceleration and in cruise. In the case of the A-60 and A-62 designs the cruise boom level is comparable, since although the latter aircraft is some 20% heavier than the Mach 2.2 design this is offset by the increase of mean cruise altitude from 60,000 ft. to 72,000 ft. The estimated ground pressure level is 1.5 to 1.6 lb/sq. ft. in each case, and this is approximately equivalent to the design level suggested by N.A.S.A. It is not unreasonable to conclude that difficulties are likely to arise with M = 2.2 and M = 3.0 designs which are heavier than the A-60 and A-62 respectively, and this may well limit the payload which can be carried by a supersonic airliner. The transonic acceleration ground boom level is a serious problem and exceeds the suggested acceptable level for both the project study aircraft. It would seem, therefore, that it is very likely that transonic acceleration would have to be carried out over uninhabited regions. This difficulty can only be overcome by reducing the weight of the aircraft or increasing the engine thrust so that the transonic phase occurs at higher altitude. Neither alternatives are likely to be economically acceptable.

11.5 Operational Considerations

The high altitude, high speed flight of supersonic airliners introduces some novel operational problems. Both the M = 2.2 and M = 3.0 designs cruise at altitudes in excess of 50,000 ft. and rapid decompression of the cabin would be fatal to the occupants. A Mach 2.2 aircraft cruises at a lower altitude than a Mach 3.0 one and in an emergency can descend to a safe altitude more rapidly. In either case the most likely cause of dangerous decompression is a door blow out so that it is essential for the design to be fail safe in this respect.

Neither cosmic radiation or ozone are likely to be hazards to a Mach 2.2 airliner cruising at around 60,000 ft., but they may well be a serious matter in the case of a Mach 3.0 design which will cruise at altitudes approaching 80,000 ft. The ozone problem can be overcome by a sealed cabin design as proposed for the A-62 project, but the radiation hazard may make it necessary to impose restrictions upon the crew flying time at these high altitudes.

The relatively modest cruise equilibrium temperature of about 130°C for the Mach 2.2 airliner does not impose any serious difficulties on the type of fuel, which can be used as a heat sink. On the other hand the Mach 3 cruise temperature of 280°C rules this out and special fuels will have to be developed.

11.6 Economics

A true economic comparison between Mach 2.2 and Mach 3.0 designs is at the same time both of vital importance and extremely difficult to achieve. However the direct comparison of the A-60 and A-62 in this respect reveals some interesting facts. Both of these designs have identical payload and range performance, the maximum passenger capacity being 120 in each case. Leaving aside the question of utilisation it would seem by comparison with present aircraft costs that on a productivity basis an airline would be prepared to pay about £4 million for the $M = 2.2$ aircraft and £5.3 million for the Mach 3.0 design. The estimation of production and development costs is an almost impossible task, but the extra complication associated with the faster aircraft probably means that it will cost at least 50% more to produce than its slower counterpart. In this respect therefore the Mach 2.2 design would seem to be preferable.

The likely utilisation of supersonic airliners is a crucial point in any economic argument. The extra complication, especially in the $M = 3.0$ aircraft would suggest that, at least initially, utilisation of the order of 3,000 hours per year will be difficult to achieve. There is, however, a further aspect of this issue which may well prove to be an overriding one. It is probable that night take offs, landings, and overland flights by supersonic airliners will not be allowed. The effect of this depends in detail upon the routes involved but in general it does imply a severe restriction upon utilisation. It has been estimated that in the case of aircraft on transatlantic routes, having a turn round time of one hour and not being allowed to take off or land between 11 p.m. and 7 a.m., the maximum utilisation for a Mach 2.2 design is 2,500 hours per year and that for a Mach 3.0 aircraft 2,000 hours per year. On this particular route the Mach 2.2 arrivals and departures coincide with the times of peak traffic density, whilst the Mach 3.0 ones happen to be somewhat inconvenient.

Fuel usage accounts for about 50% of the direct operating costs in both cases. On the basis of the first costs and utilizations mentioned above the full capacity direct operating costs over 3,000 n. miles range were estimated to be 1.54 and 1.70 pence per passenger n. mile for the A-60 and A-62 designs respectively. Using similar assumptions the direct operating cost of a typical subsonic airliner was estimated to be just under 1.50 pence per passenger n. mile. Economically, therefore the A-60, Mach 2.2 design is a more viable proposition than the A-62. Mach 3.0 aircraft, but it must be emphasised that this is primarily due to the lower assumed utilisation.

12.0 Comparison of the A-60 and A-62 Designs

The comparison made between the M = 2.2 and M = 3.0 design studies enables the following conclusions to be reached.

1. Aerodynamically there is little difference in the severity of the problems encountered.

2. The design of the propulsion system for the Mach 3.0 aircraft is more critical and complex than is the case for a Mach 2.2 design.

3. A Mach 3.0 aircraft introduces a serious materials problem in that suitable non-metallics are not available and the use of steel or titanium as the primary airframe material implies added manufacturing difficulties.

4. There is little to choose between the designs from the point of view of the cruise supersonic boom pressure which is experienced at ground level. The transonic acceleration ground pressure is more severe for the Mach 3.0 aircraft but even that due to the Mach 2.2 design is too high for operation over inhabited regions.

5. The higher altitude flight of the Mach 3.0 aircraft implies more severe operational problems, especially those arising from sudden cabin decompression, cosmic radiation and ozone poisoning.

6. Economically the M = 2.2 design is slightly perferable, but only on the basis of a utilisation determined by restrictions being placed upon night operations.

7. In general the Mach 2.2 design represents a logical step forward from current practice and its development introduces far fewer difficulties than does a Mach 3.0 aircraft. In spite of the lower cruising speed the Mach 2.2 aircraft is comparable economically and seems to be better suited to take advantage of traffic conditions. Therefore, of the two designs considered it is concluded that the Mach 2.2 aircraft is the more promising one for the immediate future.

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TABLE 1A

WEIGHT BREAKDOWNS

COMPONENT	A - 60 (M = 2.2) -lb.			A - 62 (M = 3.0) -lb.		
	PREDICTED	%	ESTIMATED	PREDICTED	%	ESTIMATED
Wings	22000	7.8	23300	36600	9.4	36300
Fuselage	22100	7.8	22000	24100	6.2	24500
Foreplane	-	-	-	4700	1.2	4100
Fin	6200	1.9	6400	5000	1.3	5600
Main Undercarriage	12000	3.7	12700	15000	3.8	16000
Nose Undercarriage	2300	0.7	2500	2500	0.6	1900
Structure	64600	19.9	66900	87900	22.5	88400
Engines	28600			25000		24700
Intakes	6600			7000		5300
Jet Pipes	5700			6600		7300
Installation	-	-	-	3400		3500
Power Plant	40900	12.6	43500	42000	10.8	40800
Fuel System	4000		3500	4500		3900
Flying Controls	2400			3000		
Power Supplies	9600			10000		
Fire Protection	2000			1000		
De-icing	1400			-	-	-
Instruments	600			1000		
Radio and Radar	1000			1000		
Cabin Systems	2518		2200	8000		4300
Furnishings and Seats	8750			10000		
Systems, etc.	32268	9.9		38500	9.9	
Crew	1232			1250		
Liquids	1000			3000		
Total Equipped	140000	43.0		172650	44.3	

TABLE 1B

WEIGHT BREAKDOWNS

COMPONENT	A - 60 (M = 2.2) -lb.			A - 62 (M = 3.0) -lb.		
	PREDICTED	%	ESTIMATED	PREDICTED	%	ESTIMATED
Structure	64600	19.9	66900	87900	22.5	88400
Power Plant	40900	12.6	43500	42000	10.8	40800
Systems	32268	9.9	-	38500	9.9	
Total Equipped	140000	43.0		172650	44.3	
Passengers (108)	15600			15600		
Baggage	7100			7100		
Payload	22700	7.0		22700	5.7	
Total Zero Fuel	162700	50.0		195350	50.0	
Fuel - Start and Taxi	2000			4000		
Take off	30000					
Climb				35350		
Cruise	94000			115000		
Let down	3000			3500		
Diversions, etc.	21000			33600		
Reserve	2300			3550		
Total Fuel	162300	50.0		195000	50.0	
A. U. W.	325000	100.0		390350	100.0	

APPENDIX A

Allocation of Components for A-60 Study

Breckell, T. H.	Engine Installation
Corkell, I.	Rear Fuselage Structure
Dodd, A. J.	Forward Wing Structure
Douglas, T. J.	Fin and Rudder Structure
Eggleston, B.	Passenger Cabin Structure
Garside, J. F.	Landing Gear
Gopal, C. S.	Cabin Air Conditioning System
Gray, B. D.	Main Wing Structure
Isaaks, D. J.	Fuselage Nose and Crew Cabin Structure
Nayler, G. H. F.	Elevator Structure
Quicke, D. C.	Air Intakes
Thomas, M. G.	Aileron Structure
Turner, M. E.	Fuselage Inter-Cabin Structure
Wheeler, D. J.	Fuel System.

APPENDIX B

Allocation of Components for A-62 Study

Catlin, G.	Horizontal Control Surfaces
Crosse, P. M.	Moving Wing Tip
Gresswell, D. J.	Fuselage in way of Foreplane
Lowe, G. A.	Fin and Rudder
Mellers, B.	Cabin and Centre Fuselage
McHarrie, J. C.	Foreplane
Monk, D. E.	Engine Installation
Myers, R. V.	Main Landing Gear
Quartermaine, R. W.	Fuel System
Reid, N.	Nose Fuselage
Sarin, R. K.	Main Wing Structure
Smith, L.	Nose Landing Gear
Suiter, B. E.	Rear Fuselage and Forward Wing
Whymark, S. A.	Cabin Environment Control System
Wilkie, D. G. M.	Flying Control System.

APPENDIX C

Specification for A-60

1.0 Powerplants

Type:- 6 Bristol Siddeley Olympus Type 591 turbojets.
Intakes:- Variable area, two-dimensional type (see Figure 13)
Mounting:- Rear fuselage, buried.

2.0 Geometry

2.1	Wing	
	Gross Area, total forward of trailing edge	5500 sq. ft.
	Area of basic wing, bounded by basic delta	5350 sq. ft.
	Span	77.0 ft.
	Aspect Ratio	1.10
	Nominal leading edge sweepback	73°
	Trailing edge sweepback	0°
	Root chord length of basic wing	133.0 ft.
	Aerodynamic mean chord	88.8 ft.
	Standard mean chord	69.5 ft.
	Equivalent taper ratio	0.05
	Wing setting angle to body datum	0°
	Aerofoil section - see Figure 8.	
2.2	Ailerons.	
	Type:- Sealed, round nose plain flaps	
	Inboard end of aileron from aircraft centreline	22.5 ft.
	Span of each aileron	16.25 ft.
	Chord at the inboard end	8.0 ft.
	Chord at 35.25 feet from centreline	6.0 ft.
	Movement (maximum)	20.5° up 20.5° down
2.3	Elevator	
	Type:- Sealed, round nose, plain flap	
	Inboard end of elevator from aircraft centreline	7.0 ft.
	Span of each elevator	15.0 ft.
	Chord	8.0 ft.
	Movement (maximum)	30.0° up 12.0° down
2.4	Fin	
	Location of fin root trailing edge, aft of wing trailing edge datum	6.70 ft
	Area, above fuselage	480 sq. ft.
	Height above fuselage datum	28.4 ft.

	True height, above fuselage	20.0 ft.
	Aspect ratio, based on true height and area	0.83
	Leading edge sweepback	66.5^o
	Trailing edge sweepback	16.0^o
	Root chord length, basic	44.0 ft.
	Aerodynamic mean chord	29.5 ft.
	Standard mean chord	24.3 ft.
	Equivalent taper ratio	0.105
	Aerofoil section :- Biconvex, 4% thickness chord ratio (Maximum thickness at 52.5% chord, with blunt trailing edge).	
2.5	Rudder	
	Type:- Sealed, round nose plain flap	
	Height, true	18.0 ft.
	Sweepback of hinge line	24.0^o
	Root chord	6.0 ft.
	Tip chord	3.0 ft.
	Movement (maximum)	$\pm 25.0^o$
2.6	Fuselage	
	Length	161 ft.
	Location of wing trailing edge datum forward of rear face of fuselage	6.7 ft.
	(Overall length of aircraft is 166 ft.)	
	Maximum depth in cabin region	10.1 ft.
	Maximum width	14.0 ft.
2.7	Powerplants	
	Location of forward face of upper engine from trailing edge datum	24.5 ft.
	Location of forward face of lower engine	22.7 ft.
	Height of upper engine centreline from fuselage datum	6.5 ft.
	Height of lower engine centreline	2.0 ft.
	Distance of outer engine centrelines from aircraft centreline	4.0 ft.
	Location of top of fuselage above engines, from datum	9.0 ft.
2.8	Undercarriage	
	Type:- Nosewheel	
	Wheelbase, to centre of bogie	62.0 ft.
	Wheelbase, bogie trimming jacks locked	64.25 ft.
	Track to centre of bogie	19.2 ft.
	Design vertical velocity	12. f.p.s.
	Ground line to fuselage datum, in static A.U.W. condition	15.5 ft.
	Main undercarriage units (8 wheel bogie):-	
	Bogie track - inboard wheels	1.8 ft.
	outboard wheels	3.8 ft.
	Bogie wheelbase	4.5 ft.
	Tyres:- 39.0 inches diameter x 11.5 inches width	

Tyre pressure	130 p.s.i.
Static tyre closure (A.U.W.)	0.1 ft.
Maximum tyre closure	0.3 ft.
Centre of bogie wheelbase forward of trailing edge	48.1 ft.
Maximum proof reaction factor	1.5
Nosewheel units (Twin wheels):-	
Track	1.1 ft.
Tyres:- 48.0 inches diameter x 8.0 inches width	
Tyre pressure	150 p.s.i.
Static tyre closure (A.U.W.)	0.07 ft.
Maximum tyre closure	0.21 ft.

3.0 Weights, Centres of Gravity and Moments of Inertia

3.1

Maximum all up weight	325,000 lb.
Normal take off weight	323,000 lb.
Maximum landing weight	190,000 lb.
Minimum landing weight	165,000 lb.
Maximum fuel load	162,300 lb.
Normal payload	22,700 lb.

3.2 Centre of Gravity positions (Zero fuel).

a. Undercarriage extended — 51.0 ft. forward of trailing edge datum; 0.4 ft. above fuselage datum

b. Undercarriage retracted — 51.8 ft. forward of trailing edge; 1.2 ft. above fuselage datum

Fuel is used to adjust centre of gravity position, the nominal location at all up weight being 54.0 ft. forward of the trailing edge and 0.7 ft. above the fuselage datum, when the undercarriage is retracted.

3.3 Moments of Inertia

Pitch	2.3×10^8 to 2.7×10^8 lb. ft.^{2} for weights from 163,000 - 323,000 lb.
Yaw	2.5×10^8 to 3.15×10^8 lb. ft.^{2} for weights from 163,000 - 323,000 lb.
Roll	0.20×10^8 to 0.52×10^8 lb. ft.^{2} for weights from 163,000 - 323,000 lb.

4.0 Aerodynamic Data

4.1 General Information

Maximum usable lift coefficient (untrimmed) (17° incidence limitation)	0.55
Normal approach lift coefficient	0.47
Drag polar (cruise at M = 2.2)	$C_D = 0.0074 + 0.412C_L{}^2$
Drag polar (cruise configuration, low speed)	$C_D = 0.004 + 0.39C_L{}^2$
Increment in C_{D_0} due to undercarriage	0.016
Pitching moment coefficient at zero lift (low speed)	-0.005
Location of aerodynamic centre, forward of trailing edge datum (low speed)	55.0 ft.

	Location of centre of pressure, forward of trailing edge datum ($M = 2.2$, $C_L = 0.096$)	51.8 ft.
4.2	Derivatives. Cruise configuration, low speed	
	Slope of wing-body lift curve, a_1 mean	1.8/rad
	Ground effect factor on a_1	1.1
	Slope of lift curve due to elevator angle, a_{2E}	0.38/rad
	Location of elevator incremental lift, forward of trailing edge datum	37.0 ft.
	Slope of fin lift curve, a_{1F}	2.0/rad
	Slope of lift curve due to rudder angle, a_{2F}	1.1/rad
	Rolling moment coefficient due to rolling, l_p	-0.08
	Rolling moment coefficient due to sideslip, l_v	$-(0.08 + 0.34C_L)$
	Rolling moment coefficient due to yawing, l_r	$0.9 - 0.4C_L$
	Rolling moment coefficient due to aileron angle, l_ξ	-0.096
	Yawing moment coefficient due to yawing, n_r	$-10.46 + 0.76C_L^2$
	Yawing moment coefficient due to sideslip, n_v	$0.17 + 0.18C_L^2$
4.3	Derivatives, Cruise configuration, $M = 2.2$	
	Slope of wing-body lift curve, a_1	1.45/rad
	Slope of lift curve due to aileron angle, a_{2A}, per aileron	0.04/rad
	Slope of lift curve due to elevator angle, a_{2E}	0.09/rad
	Location of elevator incremental lift, forward of trailing edge datum (with allowance for aeroelastic distortion)	14.0 ft.
	Slope of fin lift curve, a_{1F}	1.2/rad
	Slope of lift curve due to rudder angle, a_{2F}	0.34/rad
	Location of rudder incremental lift, aft of trailing edge datum	6.0 ft.

4.4 Control Characteristics

m, stick gearing, is in rad/foot.

Q is $\left(\frac{V}{100}\right)^2$

where V is equivalent flight speed, f.p.s. E.A.S.

Elevator		
Maximum booster effort		185,000 ft. lb.
Maximum control application rate		40 degrees/sec.
Stick gearing, m	$V \leq 300$	0.75
	$V \geq 300$	$0.75 - 0.0094(Q-9)$
Rudder		
Maximum booster effort		75,000 ft. lb.
Maximum control application rate		30 degrees/sec.
Stick gearing, m	$V \leq 460$	1.2
	$V \geq 460$	$1.2 - 0.0114(Q-21.2)$
Aileron		
Maximum booster effort		98,000 ft. lb.
Maximum control application rate		30 degrees/sec.
Stick gearing, m	$V \leq 245$	0.7
	$V \geq 245$	$0.7 - 0.0034(Q-6)$

5.0 Loading Requirements

The aircraft is designed to meet British Civil Airworthiness Requirements with amplification from the Military Requirements for a supersonic aircraft.

5.1 Design Envelope

The maximum unfactored normal acceleration is 2.5g associated with speeds from V_A to V_D. The design flight speeds are:-

V_S	(Maximum take off weight)	300 f.p.s. (178 knots) E.A.S.
V_S	(Maximum landing weight)	230 f.p.s. (136 knots) E.A.S.
V_A	(Maximum take off weight)	475 f.p.s. (282 knots) E.A.S.
V_C		700 f.p.s. (414 knots) E.A.S.
V_D		750 f.p.s. (445 knots) E.A.S.

(V_D is associated with a maximum Mach No. of 2.35)

5.2 Aircraft Life and Environment

The aircraft is designed to have a total life of 30,000 hours of which 20,000 hours are considered to be spent at the cruise temperature. (130°C over the greater part of the load carrying structure and 150°C at the intakes and leading edges). The 30,000 hours is assumed to be accumulated in 40,000 flights.

The maximum total creep does not exceed 0.1%.

5.3 Pitching Accelerations

The design values are:-

Speed V_A (323,000 lb.)	0.11 rad/sec²
V_C	0.077 rad/sec²
V_D	0.07 rad/sec²

5.4 Design Roll Rates

Maximum rate of roll (automatically limited) 3.20 rad/sec.

5.5 Yaw Equilibrium Angles

Speed V_A	(subsonic)	Rudder angle 25° (max.)	Yaw angle 7.0°
V_C	(subsonic)	Rudder angle 18.5° (max.)	Yaw angle 6.5°
V_D	(subsonic)	Rudder angle 16.3° (max.)	Yaw angle 5.8°
V_C	(cruise);	Hinge moment limitation	Yaw angle 2.5°
V_D	(cruise);	Hinge moment limitation	Yaw angle 2.2°

5.6 Cabin Pressurisation

The maximum cabin differential pressure is 10.5 p.s.i. (This is determined by the maximum allowable rate of descent from 65,000 ft. altitude).

5.7 Design Brake Torque

A static brake torque of 15,000 lb. ft. per brake is assumed. The energy absorption is 1.16×10^7 lb. ft. per brake on 16 brakes in the normal landing conditions.

APPENDIX D

Specification for A-62

1.0 Powerplants

Type:- 4 Bypass engines of 30,000 lb. sea level static thrust, with cold stream burning to 1150°K.
Intakes:- Variable area, axisymmetric type.
Mounting:- Individual pods below wing.

2.0 Geometry

2.1 Wing

Gross Area; tips down	4500 sq. ft.
tips up	4650 sq. ft.
Span; tips down (high speed)	102 ft.
tips up (low speed)	118 ft.
Aspect Ratio; high speed	2.31
low speed	3.0
Leading edge sweepback	50°
Trailing edge sweepforward	2.8°
Tip cut off angle	20°
Root chord length of basic wing	77 ft.
Wing tip hinge chord length	18.7 ft.
Aerodynamic mean chord; high speed	53.2 ft.
low speed	51.5 ft.
Standard mean chord; high speed	47.0 ft.
low speed	38.5 ft.
Distance of tip hinge chord from centreline	46.6 ft.
Wing angle to body datum	0°
Wing datum below fuselage datum	1.8 ft.
Movement of wing tip below wing datum	10° to 70°

Aerofoil section; 4.95% thick biconvex with maximum thickness at 0.55 chord. (Aerodynamically 4.5% on 1.10 chord)
Fore and aft wing datum passes through tips of basic wing.

2.2 Ailerons

Type:- Minimum gap round nose.

Inboard end to centreline	2.6 ft.
Outboard end to centreline	18.6 ft.
Chord at inboard end	8.7 ft.
Chord at outboard end	7.9 ft.
Sweepback of hinge line	0°
Movement	25° up 25° down

2.3 Foreplane

Area	485 sq. ft.
Span	39 ft.
Aspect ratio	3.14
Leading edge sweepback	19°
Trailing edge sweepforward	6.5°
Root chord on centreline	17.5 ft.
Tip chord	8.5 ft.
Standard mean chord	12.98 ft.
Angle of foreplane datum relative to wing datum	-0.5°
Leading edge apex forward of wing datum	145.0 ft.
Height of leading edge above fuselage datum	3.5 ft.

Aerofoil section; 6.05% thick biconvex with maximum thickness at 0.55 chord. (Aerodynamically 5.5% on 1.10 chord).

2.4 Foreplane Flap (Elevator)

Type:- Minimum gap round nose

Inboard end to centreline	5.25 ft.
Outboard end to centreline	19.5 ft.
Chord at inboard end	3.5 ft.
Chord at outboard end	2.0 ft.
Sweepback of hinge line	0°
Movement	20° up 20° down

2.5 Fin

Area outside fuselage	455 sq. ft.
Height above fuselage datum	26.8 ft.
Aspect ratio, based on true height and area	1.40
Leading edge sweepback	40°
Trailing edge sweepback	15°
Root chord length, at fuselage datum	27.0 ft.
Tip chord length	11.5 ft.
Aerodynamic mean chord	20.3 ft.
Standard mean chord, based on true height and area	19.25 ft.
Distance of fin leading edge at fuselage datum, forward of wing datum	23.5 ft.

Aerofoil section:- 4.84% thick biconvex with maximum thickness at 0.55 chord (Aerodynamically 4.4% on 1.10 chord).

2.6 Rudder

Type:- Minimum gap round nose

Area	114 sq. ft.
Height	19.3 ft.
Root chord above fuselage datum	3.1 ft.
Root chord length	7.6 ft.
Tip chord length	4.2 ft.
Sweepback of hinge line	24°
Movement	20° up 20° down

2.7	Fuselage	
	Length	205 ft.
	Length forward of wing datum	193 ft.
	Maximum diameter	13.7 ft.
	Distance of cabin floor below fuselage datum	2.5 ft.
	Height of cabin roof above fuselage datum	4.5 ft.
	Seat pitch (108 passengers) - first class	3.33 ft.
	Seat pitch (120 passengers) - high density	2.75 ft.
2.8	Powerplants	
	Pod length, excluding centrebody	29.0 ft.
	Pod maximum diameter	6.0 ft.
	Intake diameter	5.1 ft.
	Location of intake lip forward of wing datum	31.5 ft.
	Location of inboard pod from centreline	22.0 ft.
	Location of outboard pod from centreline	34.0 ft.
	Distance of inboard pod below wing datum	6.8 ft.
	Distance of outboard pod below wing datum	5.3 ft.
2.9	Undercarriage	
	Type:- Nosewheel	
	Wheelbase (to centre of bogie)	62.5 ft.
	Track (to centre of bogie)	37.0 ft.
	Design vertical velocity (proof)	12.0 ft/sec.
	Ground line below fuselage datum at A.U.W.	18.5 ft.
	Main undercarriage units (8 tyre bogie)	
	Bogie track, - inboard wheels	4.0 ft.
	Bogie track, - outboard wheels	8.0 ft.
	Bogie wheelbase	5.2 ft.
	Tyres:- 52 inches diameter x 10.4 inches width	
	Tyre pressure	140 p.s.i.
	Centre of bogie wheelbase forward of wing datum	44.0 ft.
	Maximum proof reaction factor	1.7
	Nosewheel unit (twin wheels)	
	Track	2.0 ft.
	Tyres:- 52 inches diameter x 10.4 inches width	
	Tyre pressure	100 p.s.i.

3.0 Weights, centres of gravity and moments of inertia

3.1	Maximum all up weight	390,350 lb.
	Maximum landing weight	232,500 lb.
	Minimum landing weight	198,900 lb.
	Maximum fuel load	190,000 lb.
	Normal payload (108 passengers)	22,700 lb.
	Maximum payload (120 passengers)	25,200 lb.

3.2 Centre of Gravity (Zero fuel)

a. Undercarriage extended	48.6 ft. forward of trailing edge datum. 2.95 ft. below fuselage datum
b. Undercarriage retracted	48.6 ft. forward of trailing edge datum. 1.95 ft. below fuselage datum.

Fuel is used to adjust the centre of gravity position, the nominal location being 49.6 ft. forward of the datum at all up weight, with a range of from 49.0 ft. to 51.4 ft. forward of the datum.

3.3 Moments of Inertia

Pitch 3.19×10^8 to 6.05×10^8 lb. ft.^{2} for weights from 172,650 - 390,350 lb.
Yaw 3.75×10^8 to 7.01×10^8 lb. ft.^{2} for weights from 172,650 - 390,350 lb.
Roll 0.61×10^8 to 1.04×10^8 lb. ft.^{2} for weights from 172,650 - 390,350 lb.

4.0 Aerodynamic Data

4.1 General Information

Maximum low speed lift coefficient (trimmed)	0.96
Maximum foreplane lift coefficient (zero elevator)	0.87
Normal approach lift coefficient (12° wing incidence)	0.746
Unstick lift coefficient (10.3° wing incidence)	0.64
Drag polar (cruise at M = 3.0) $C_D = 0.009 + 0.53C_L^2$	
Drag polar (cruise at low speed) $C_D = 0.011 + 0.20C_L^2$	
Pitching moment coefficient at zero lift	-0.005
Location of wing-body aerodynamic centre, forward of trailing edge datum:-	
Low speed	40.5 ft.
M = 1.0, tips extended	41.0 ft.
M = 1.0, tips drooped	51.5 ft.
M = 3.0, tips drooped	39.0 ft.
(Tips are drooped progressively from M = 0.9 to M = 1.3)	
Location of foreplane aerodynamic centre, forward of trailing edge datum:-	
Low speed	138.0 ft.
M = 3.0	135.5 ft.

4.2 Derivatives; cruise configuration

Slope of wing-body lift curve, a_1;	low speed	3.2/rad
	M = 1.0	4.2/rad
	M = 3.0	1.6/rad
Slope of foreplane lift curve, a_{1f};	low speed	3.6/rad
	M = 1.0	6.0/rad
	M = 3.0	1.8/rad
Slope of fin lift curve, a_{1R};	low speed	3.3/rad
	M = 1.0	5.1/rad
	M = 3.0	1.8/rad
Ratio of inboard wing control (aileron) lift curve slope, a_2/a_1, low speed		0.092

Ratio of foreplane control (elevator) lift curve slope,

$\frac{a_2 f}{a_{1f}}$, low speed	0.38
Ratio of rudder lift curve slope, $^{a_2R}/_{a_1R}$, low speed	0.56
Rolling moment coefficient due to rolling, low speed, ℓ_p	-0.25
Rolling moment coefficient due to sideslip, low speed, ℓ_v	$-(0.03 + 0.27C_L)$
Rolling moment coefficient due to yawing, low speed, ℓ_r	$0.02 + 0.17C_L$
Yawing moment coefficient due to yawing, low speed, n_r	$-(0.07 + 0.03C_L)$
Yawing moment coefficient due to sideslip, low speed n_v	0.0676

5.0 Loading Requirements

The aircraft is designed to meet British Civil Airworthiness Requirements with amplification from the Military Requirements for a supersonic aircraft.

5.1 Design Envelope

The maximum unfactored normal acceleration is 2.5g associated with speeds from V_A to V_C. The design flight speeds are:-

V_S (maximum weight)	163 knots E.A.S.
V_S (maximum landing weight)	126 knots E.A.S.
V_A (maximum weight)	257 knots E.A.S.
V_C	477 knots E.A.S.
V_D	531 knots E.A.S.

The maximum normal operating Mach number of 3 is associated with a maximum E.A.S. of 430 knots. The design covers an overspeed to M = 3.2 for a short period, associated with an appropriate increase in E.A.S.

5.2 Aircraft Life and Environment

The aircraft is designed to have a total life of 30,000 hours of which 20,000 hours are considered to be spent at the cruise temperature (280°C over the greater part of the load carrying structure and 325°C at stagnation points). The 30,000 hours is assumed to be accumulated in 30,000 complete flights.

5.3 Pitching Accelerations

The pitching accelerations were assumed not to exceed $45/V_{E.A.S.}$ rad/sec., where V is in knots.

5.4 Cabin Pressurisation

11.3 p.s.i. is the maximum cabin differential pressure. (This is determined by assuming a maximum cabin rate of descent of 500 ft/min. during the descent of the aircraft from 76,000 ft. altitude. Cabin altitude does not exceed 6,000 ft.)

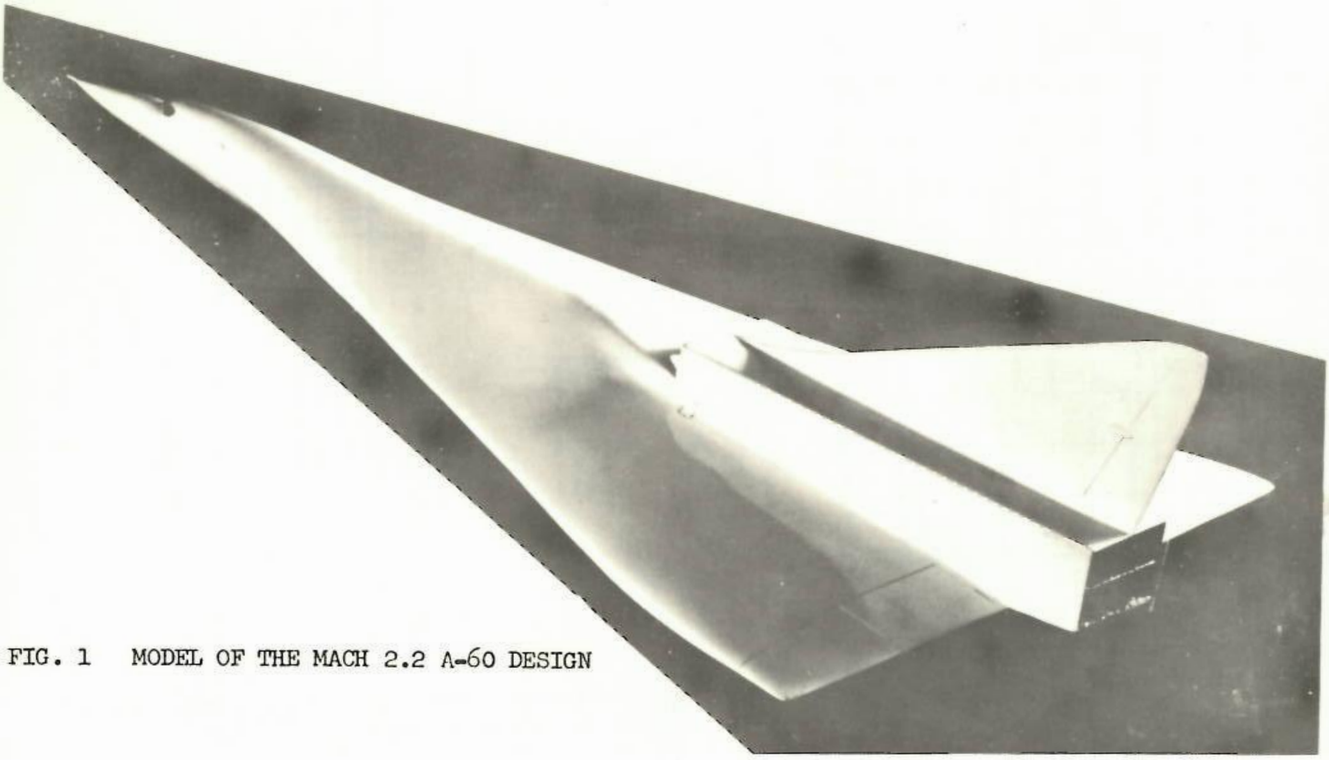


FIG. 1 MODEL OF THE MACH 2.2 A-60 DESIGN

FIG. 2 MODEL OF THE MACH 3.0 A-62 DESIGN

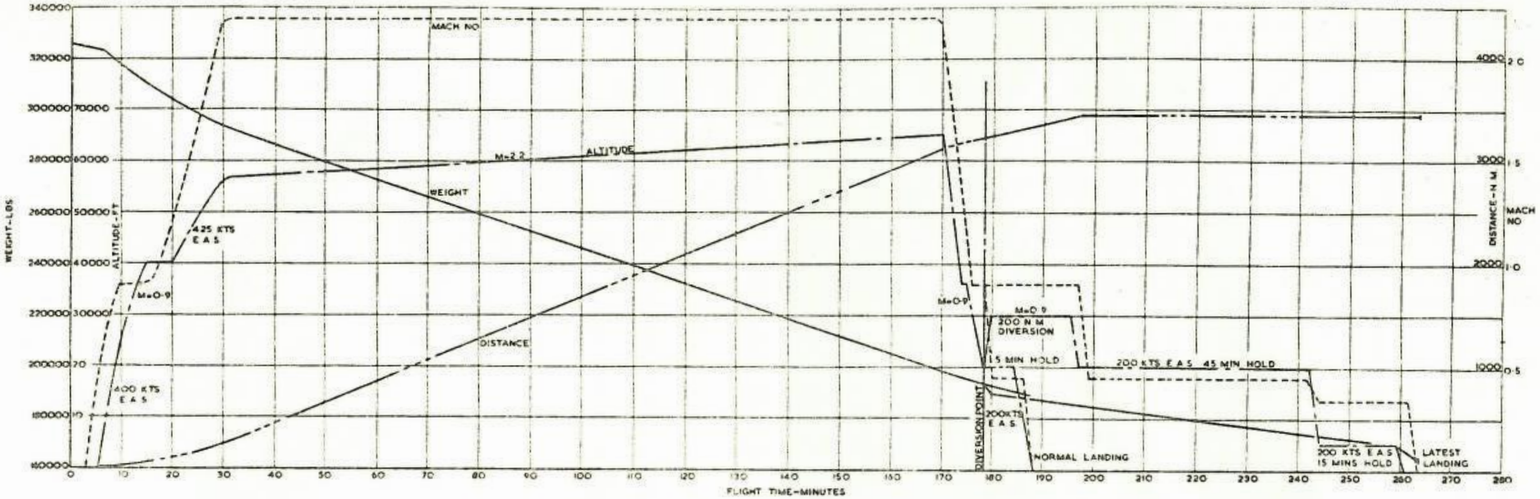


FIG. 3 TYPICAL FLIGHT PLAN FOR M = 2·2 A-60 DESIGN

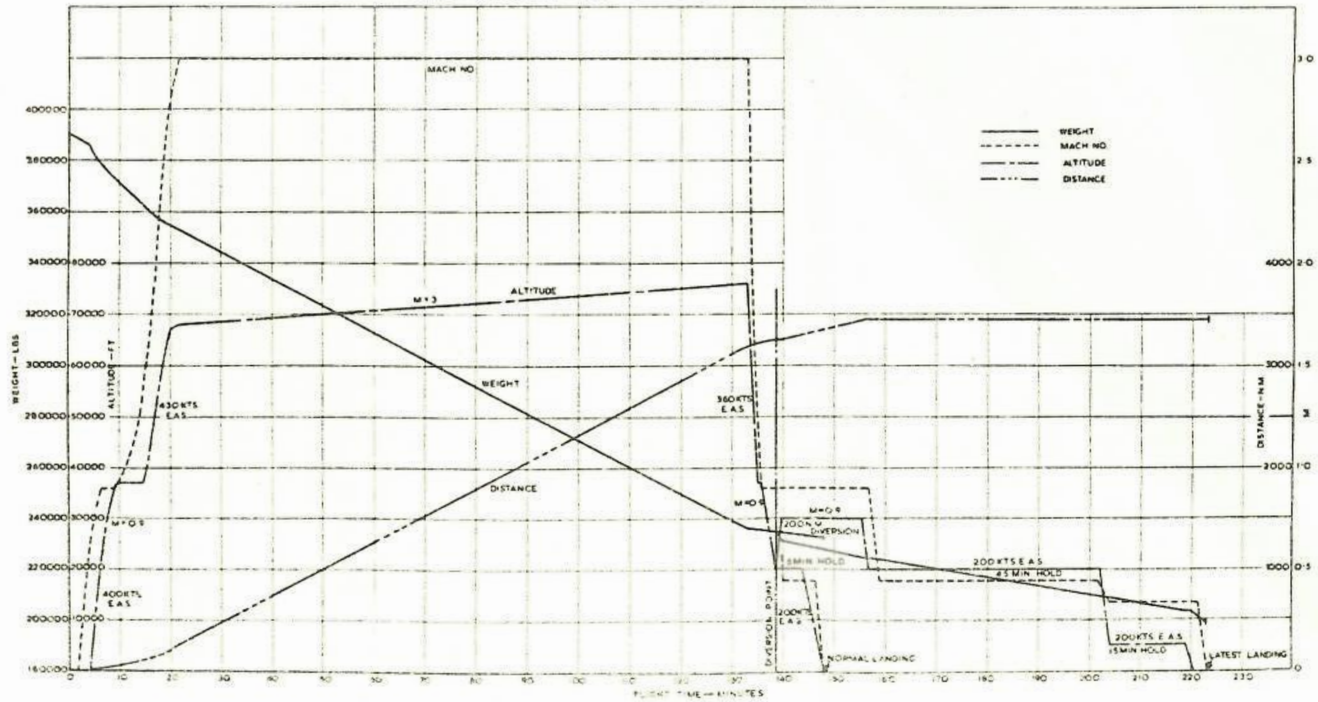


FIG. 4 TYPICAL FLIGHT PLAN FOR M = 3·0 A-62 DESIGN

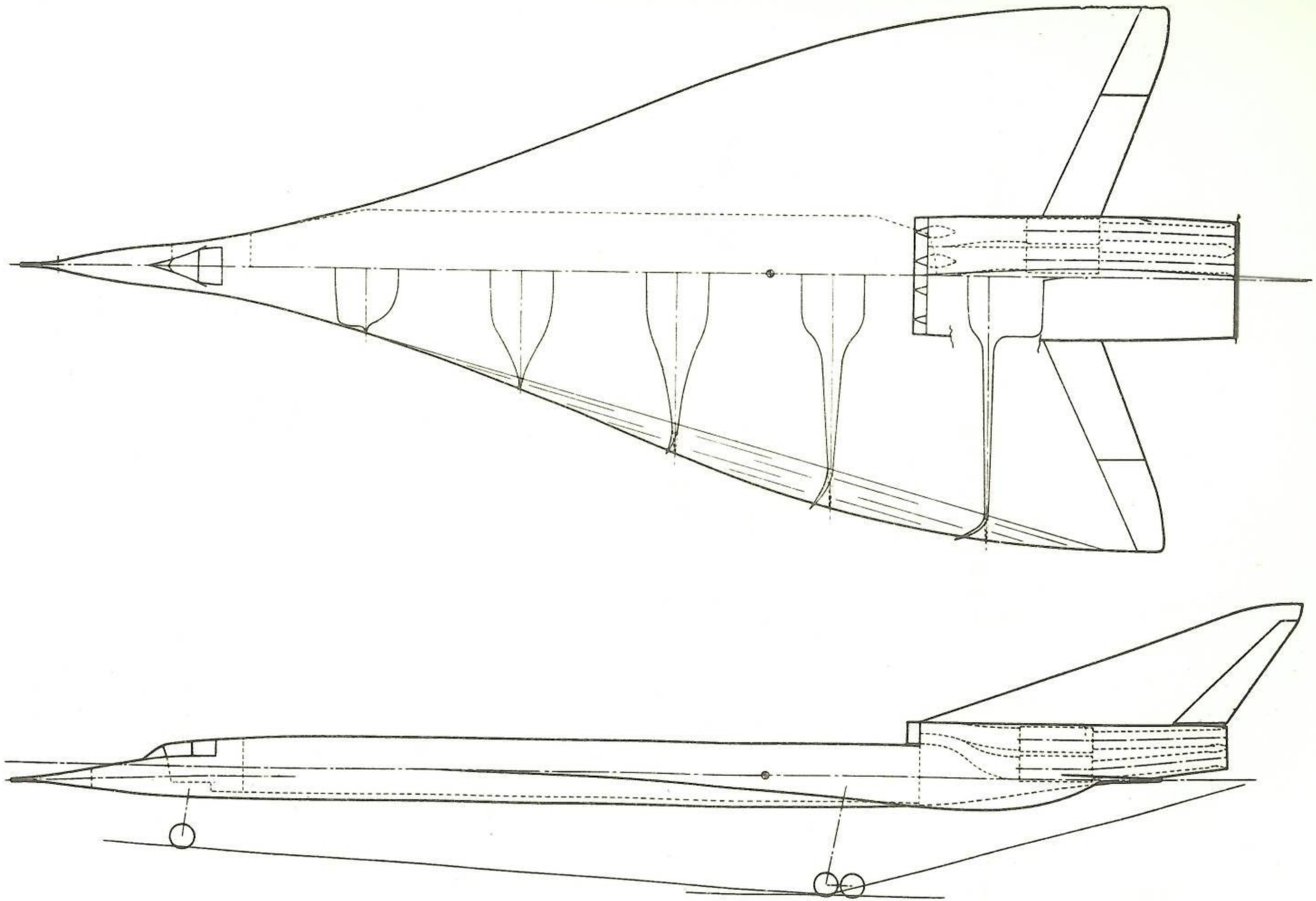


FIG.5. GENERAL ARRANGEMENT DRAWING OF PRELIMINARY M=1·8 DESIGN.

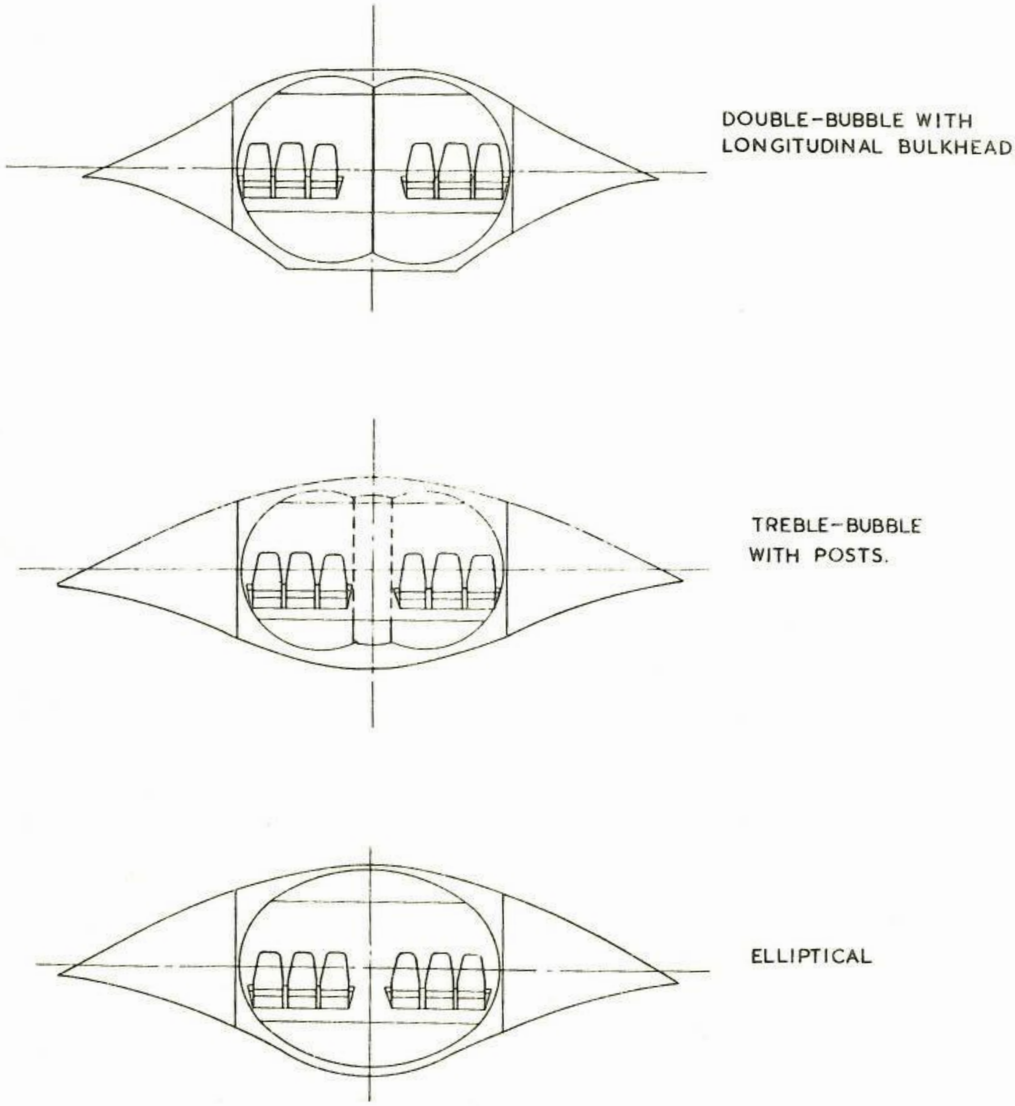


FIG.6. COMPARISON OF CABIN CROSS SECTIONS.

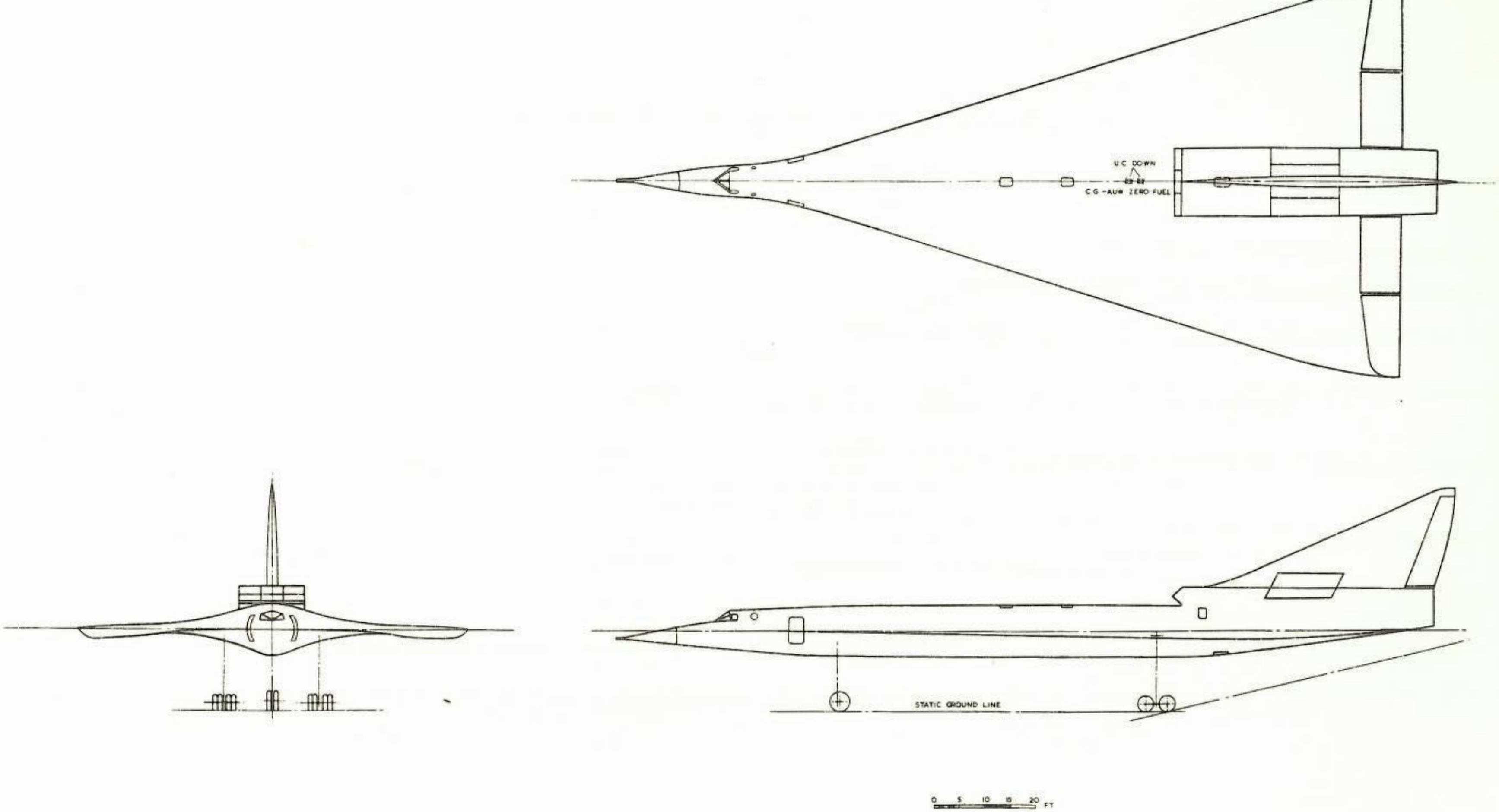


FIG.7. GENERAL ARRANGEMENT DRAWING OF M=2·2 A-60 DESIGN.

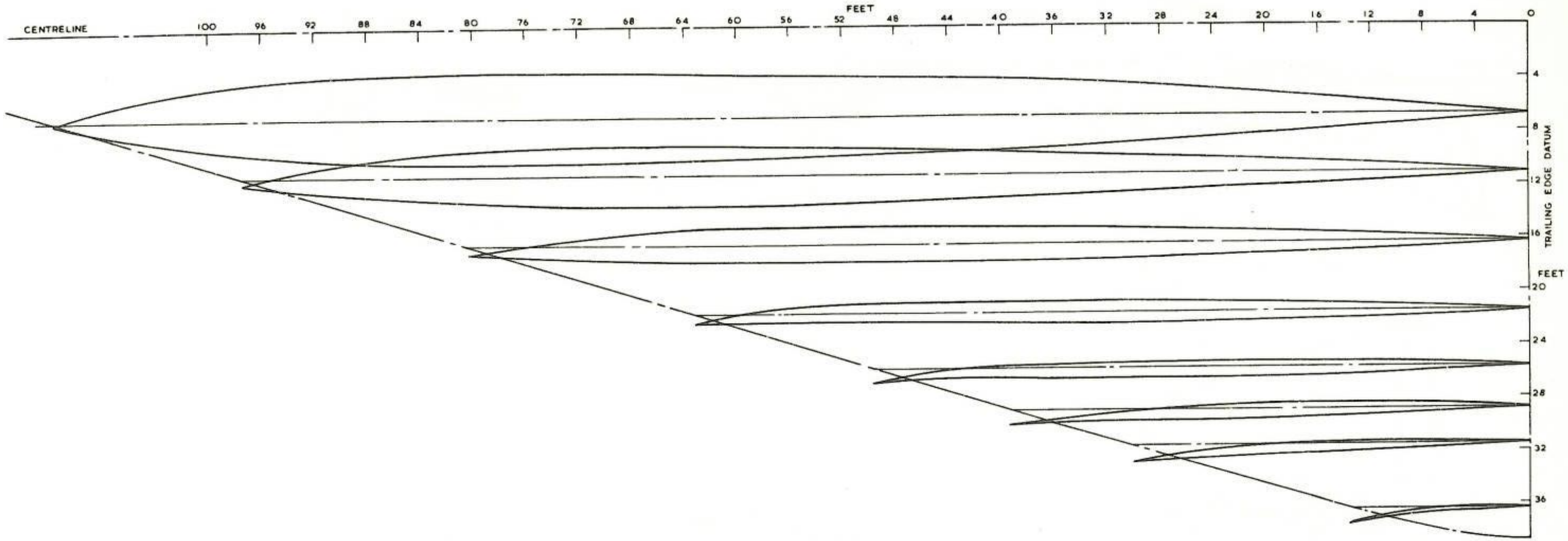


FIG.8. WING SECTIONS OF M= 2·2 A-60 DESIGN.

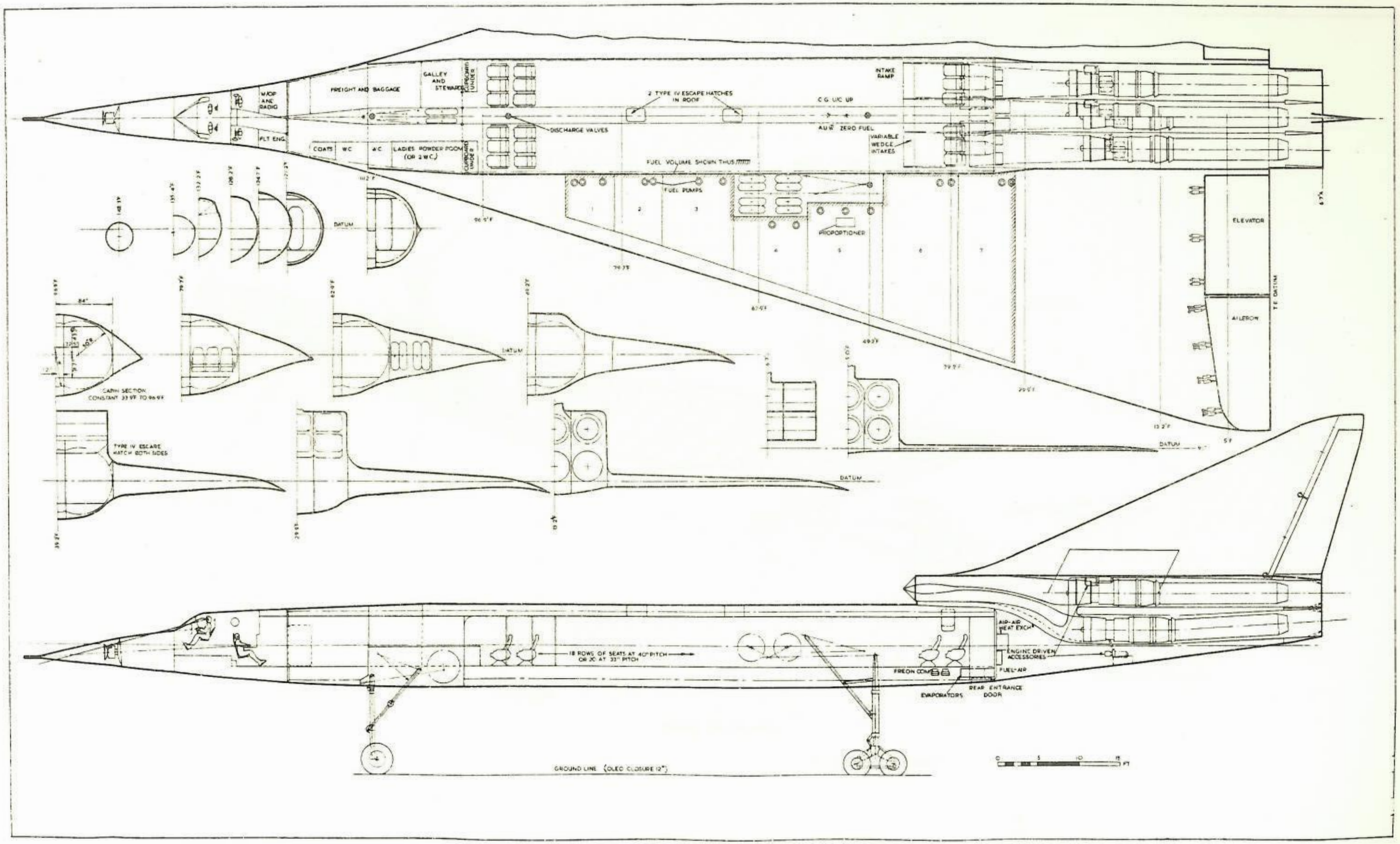


FIG.9. INTERNAL LAYOUT OF M=2·2 A-60 DESIGN.

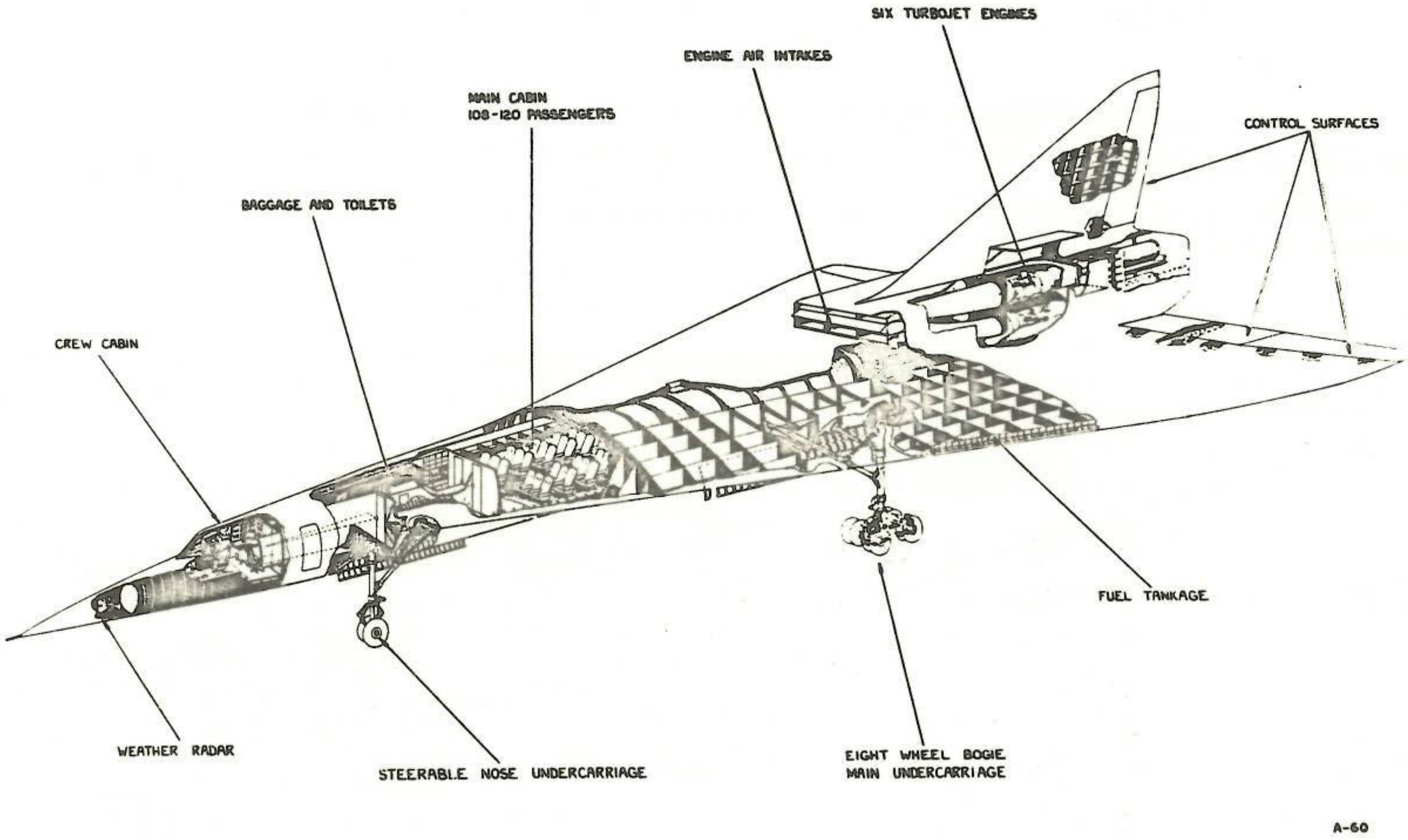


FIG. 10 CUTAWAY DRAWING OF M = 2.2, A — 60 DESIGN

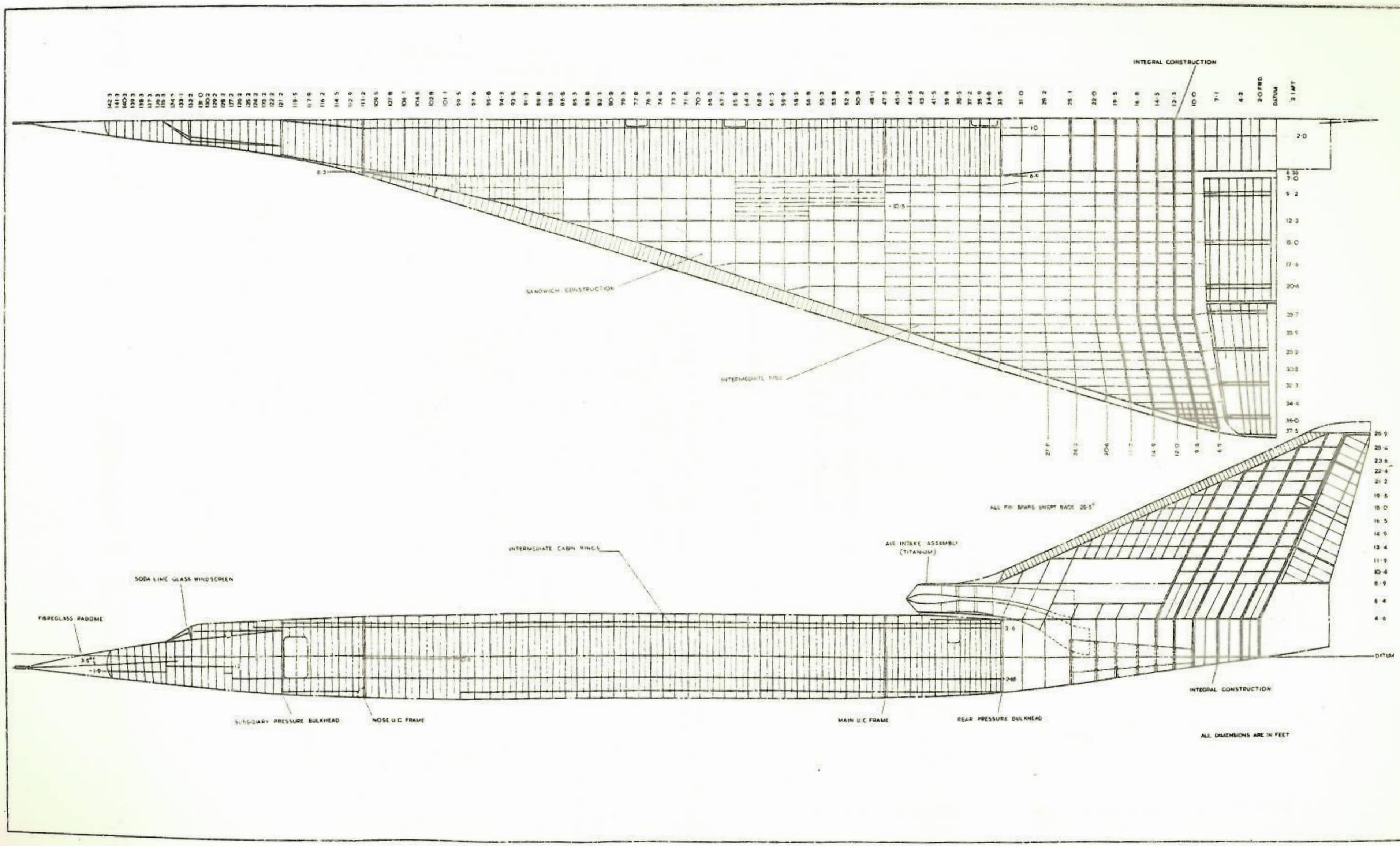
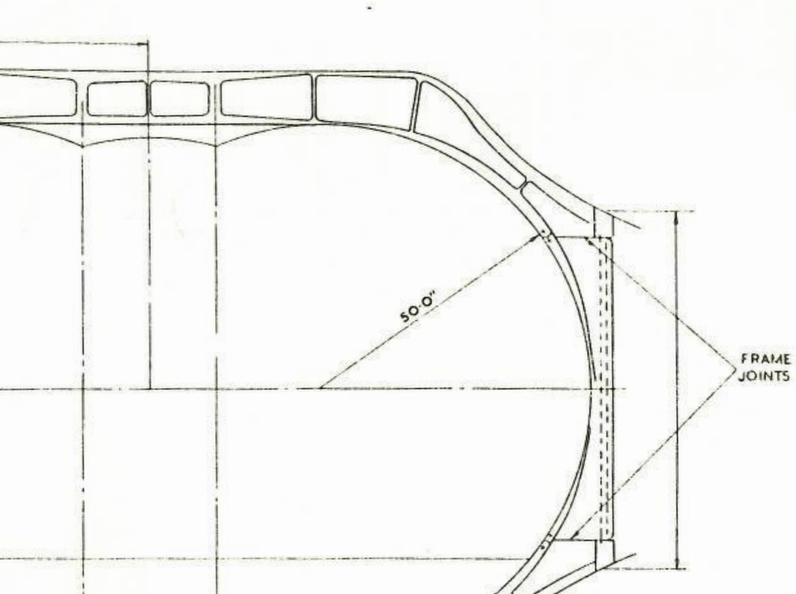


FIG.11. KEY STRUCTURAL DRAWING OF M=2·2 A-60 DESIGN.


50·0"
FRAME
JOINTS

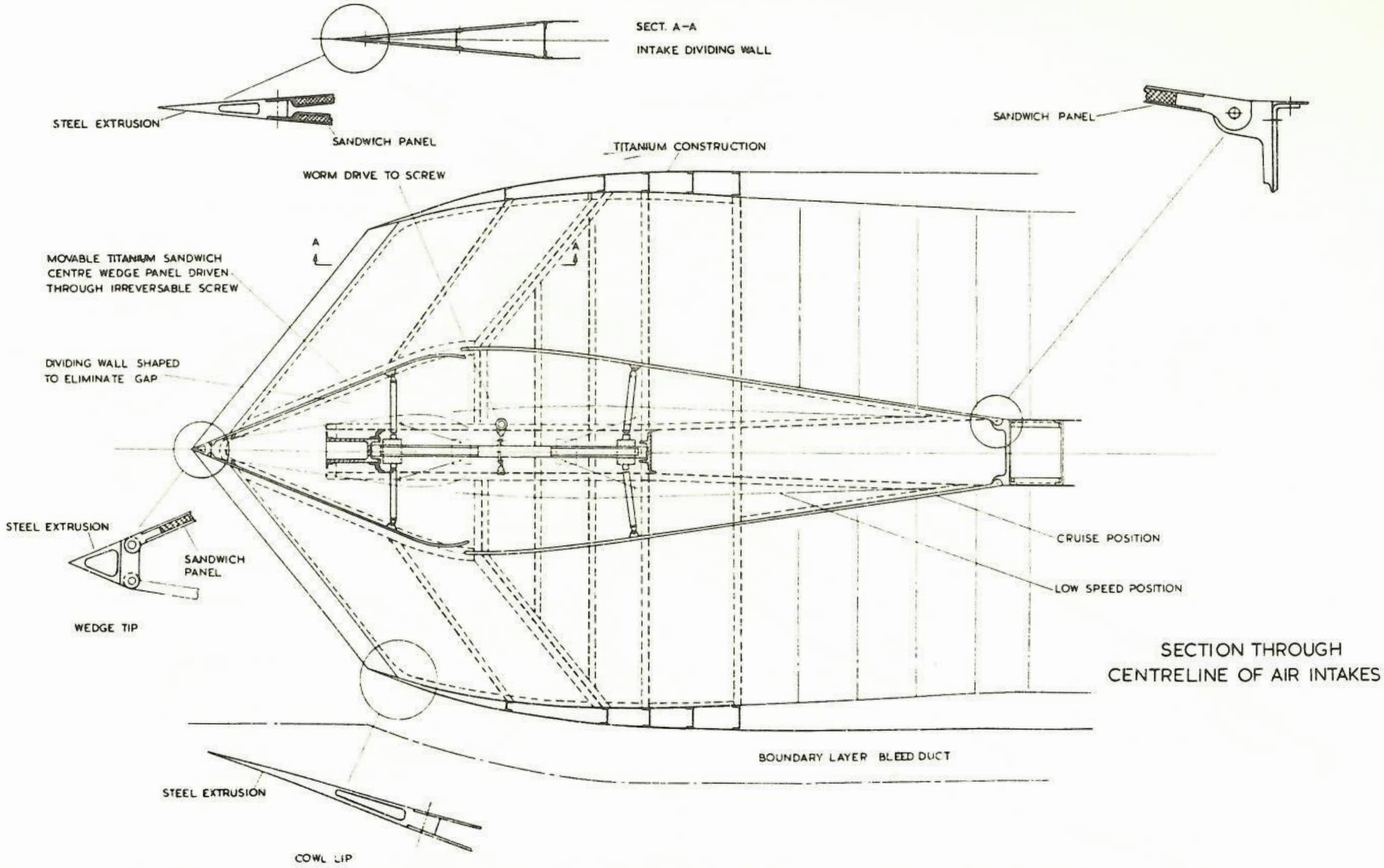


FIG.13. LAYOUT OF INTAKES OF M=2·2 A-60 DESIGN.

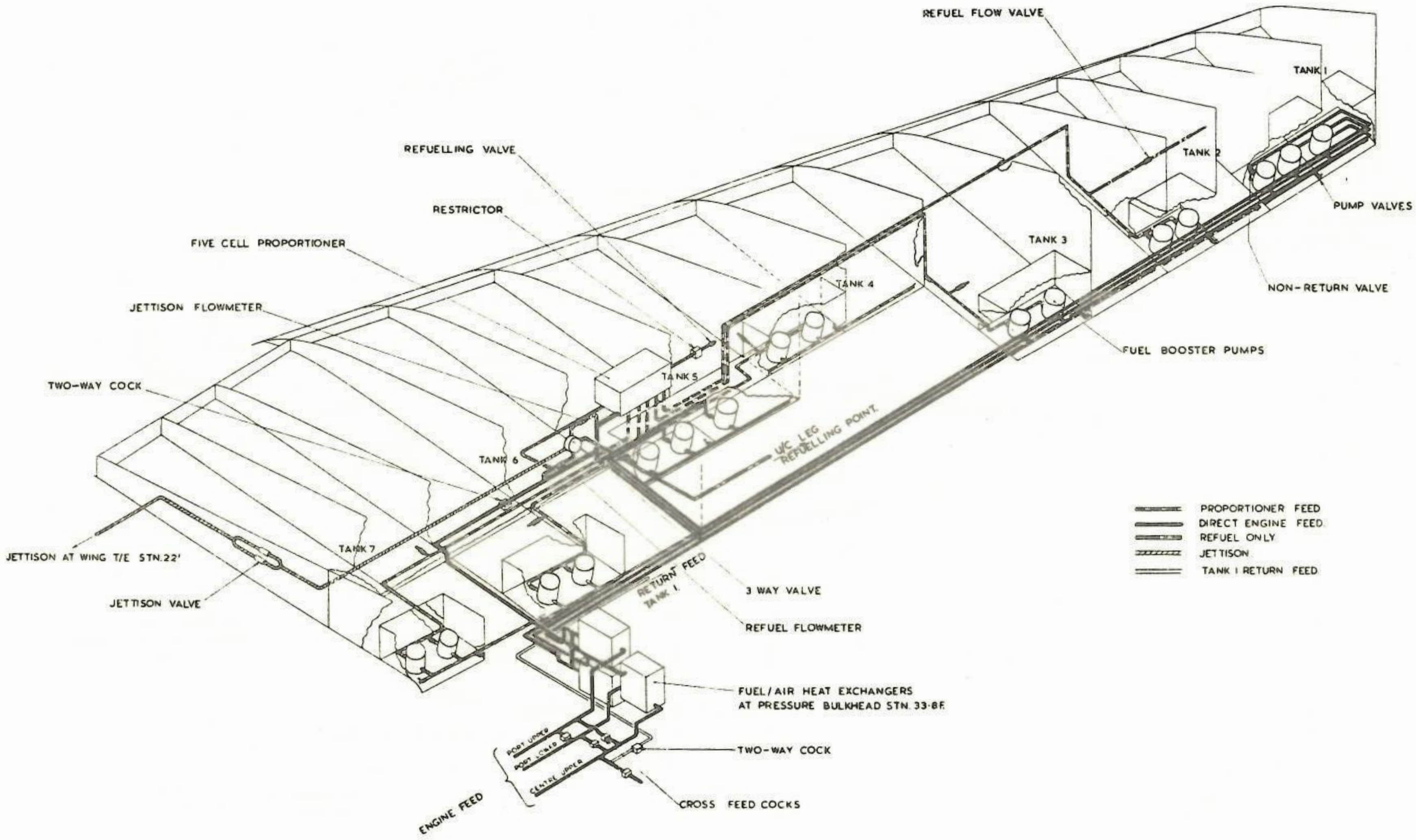


FIG.14. FUEL SYSTEM OF M=2·2 A-60 DESIGN.

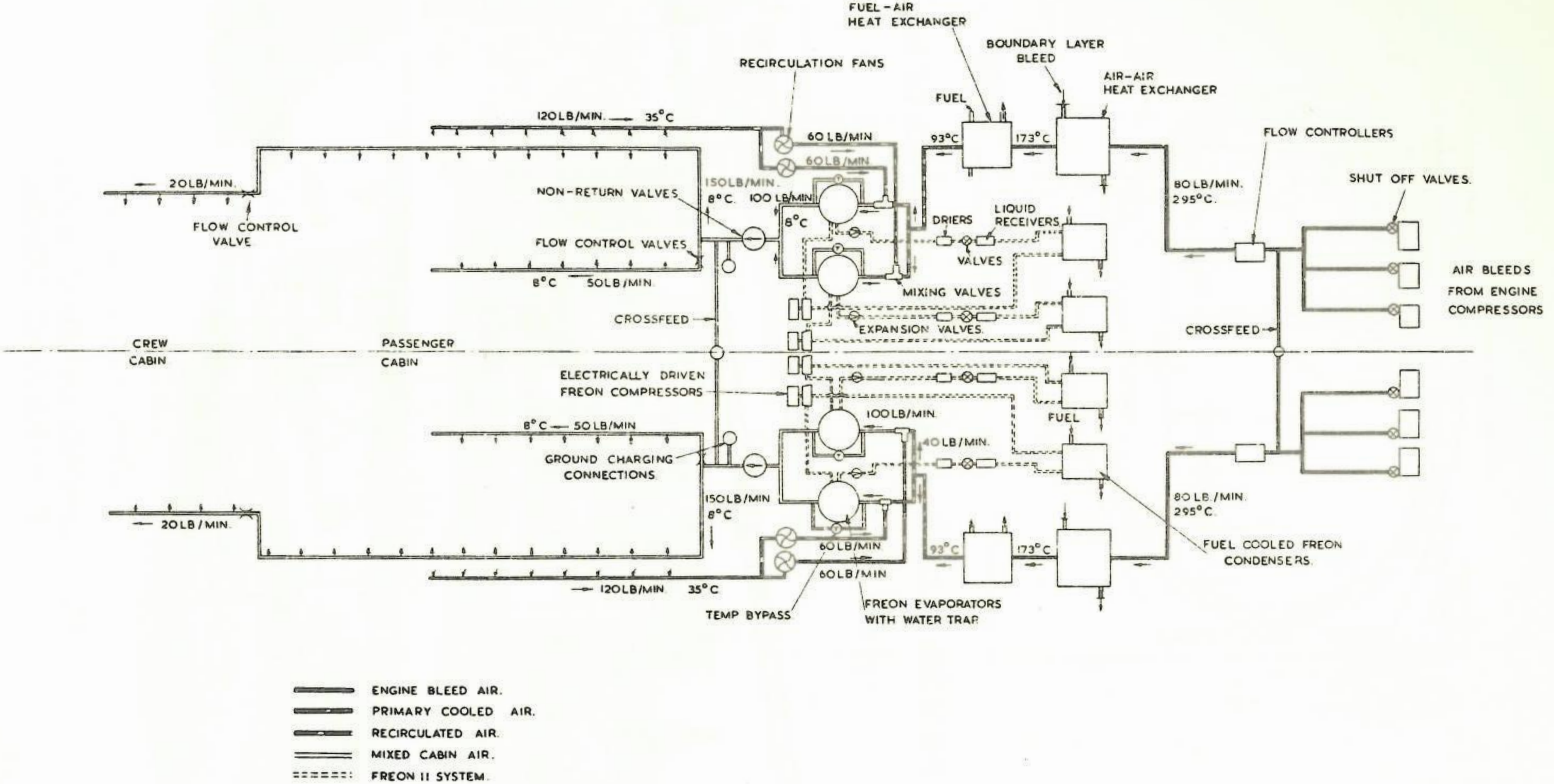


FIG.15. CABIN ENVIRONMENTAL CONTROL SYSTEM OF M=2·2 A-60 DESIGN.

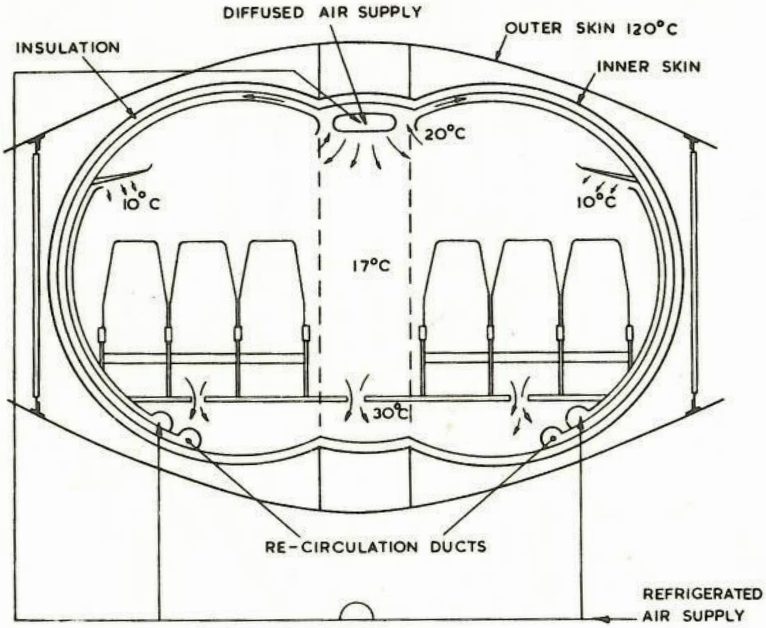


FIG.16. AIR FLOW IN CABIN OF M=2·2 A-60 DESIGN.

FIG.17. LANDING DYNAMIC MODEL
OF M=2·2 A-60 DESIGN.

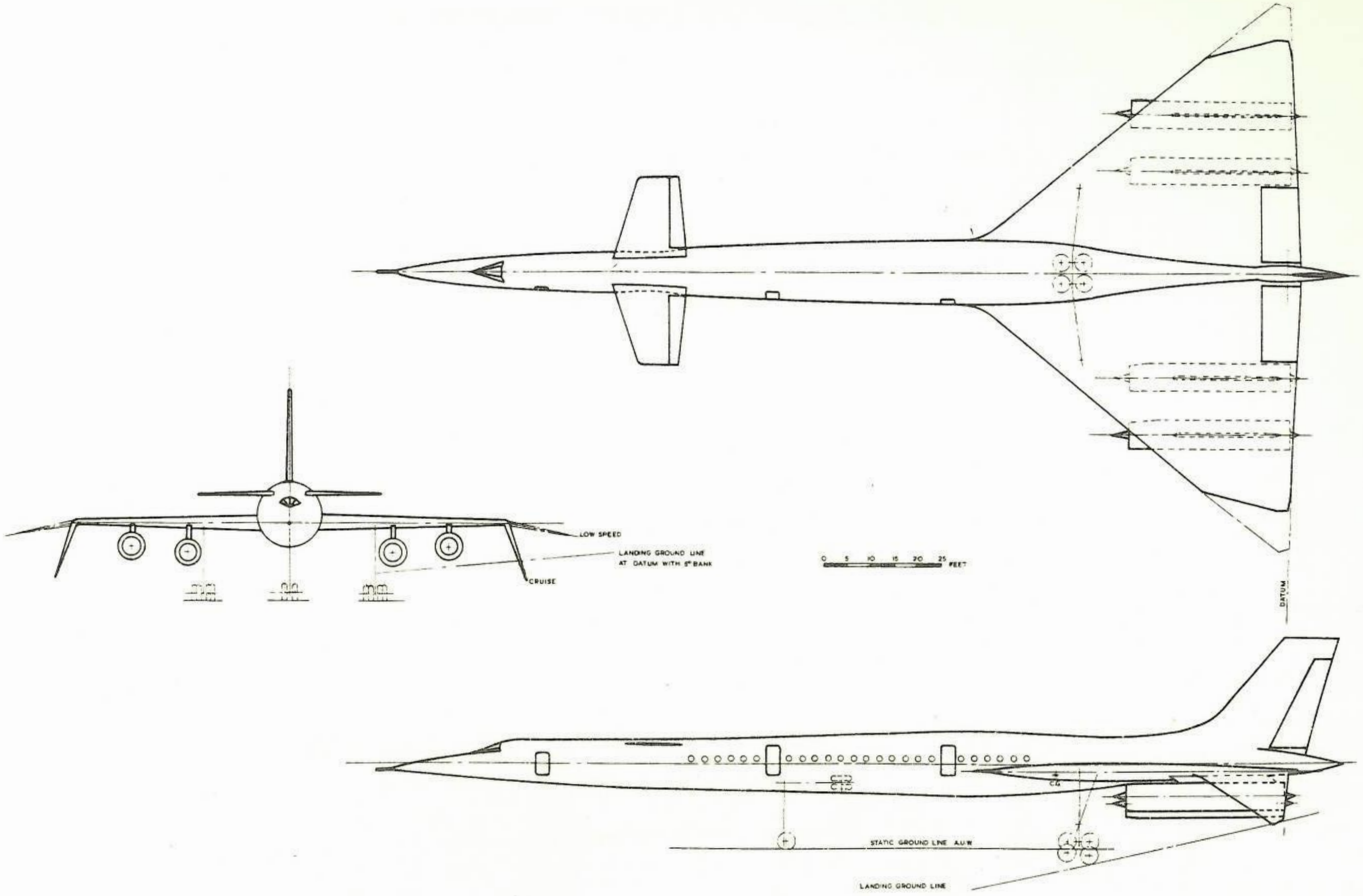


FIG.18. GENERAL ARRANGEMENT DRAWING OF M=3·0 A-62 DESIGN

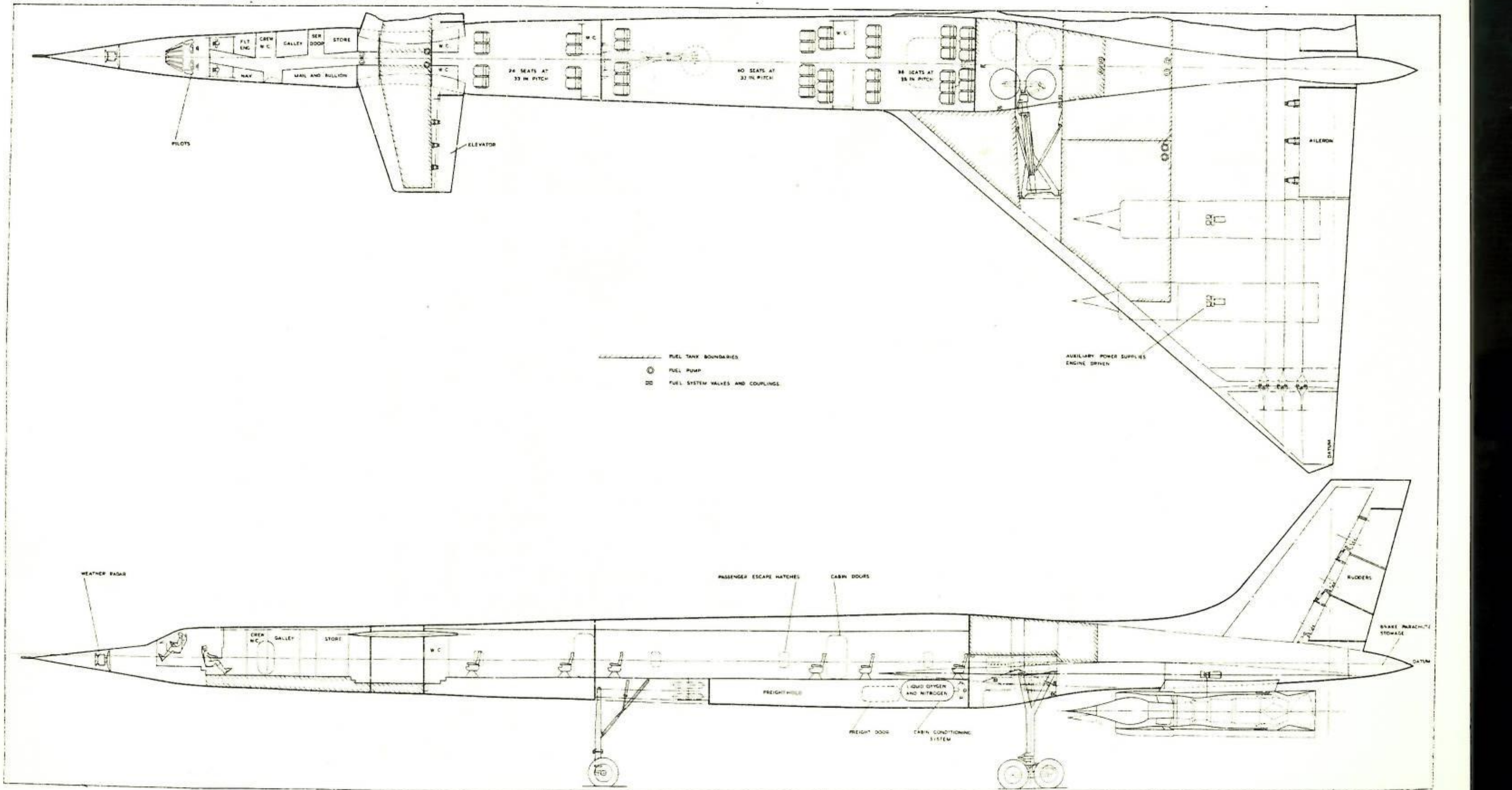


FIG.19. INTERNAL LAYOUT OF M=3·O A-62 DESIGN.

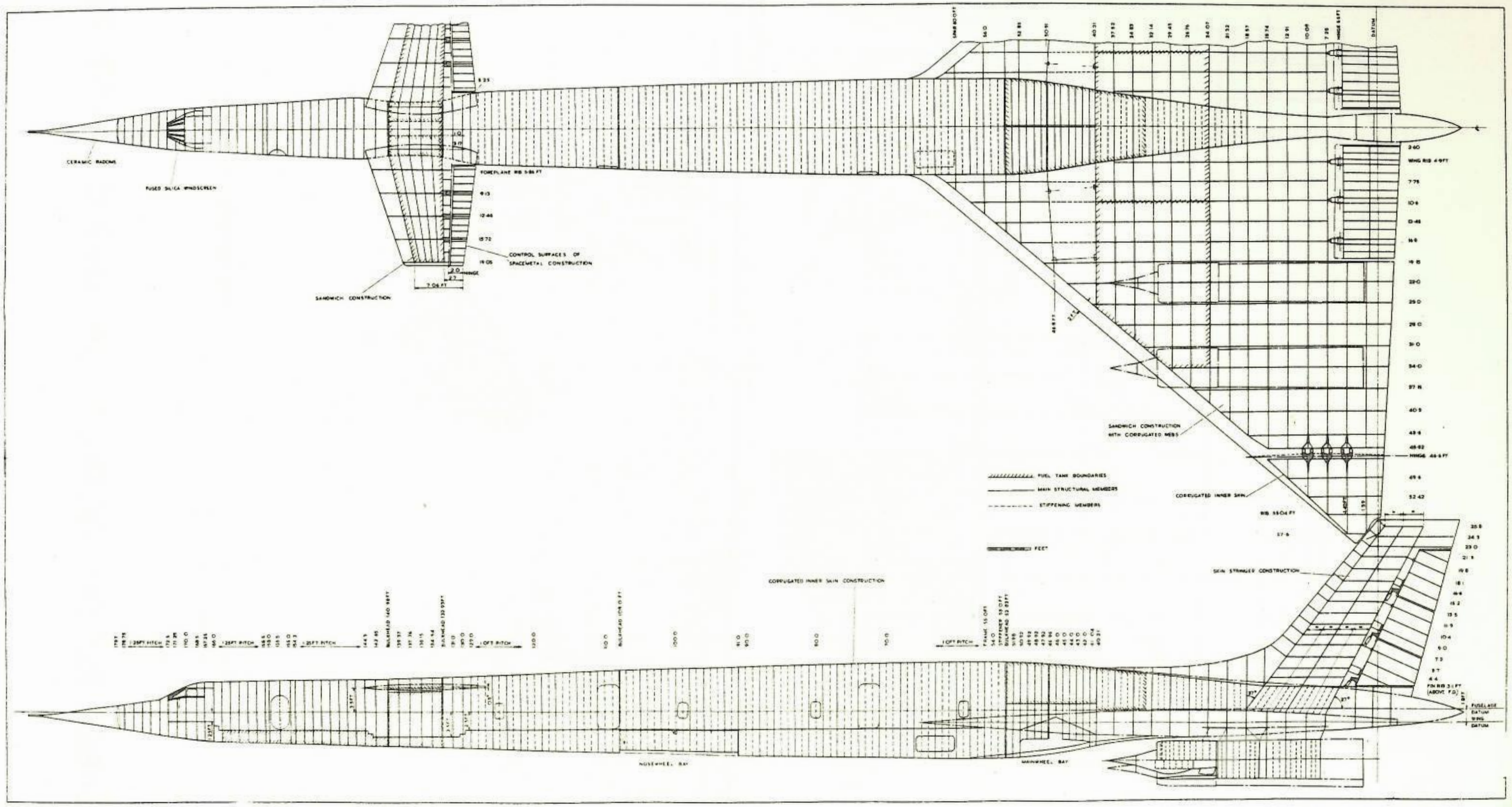


FIG.20. KEY STRUCTURAL DRAWING OF M=3·O A-62 DESIGN.

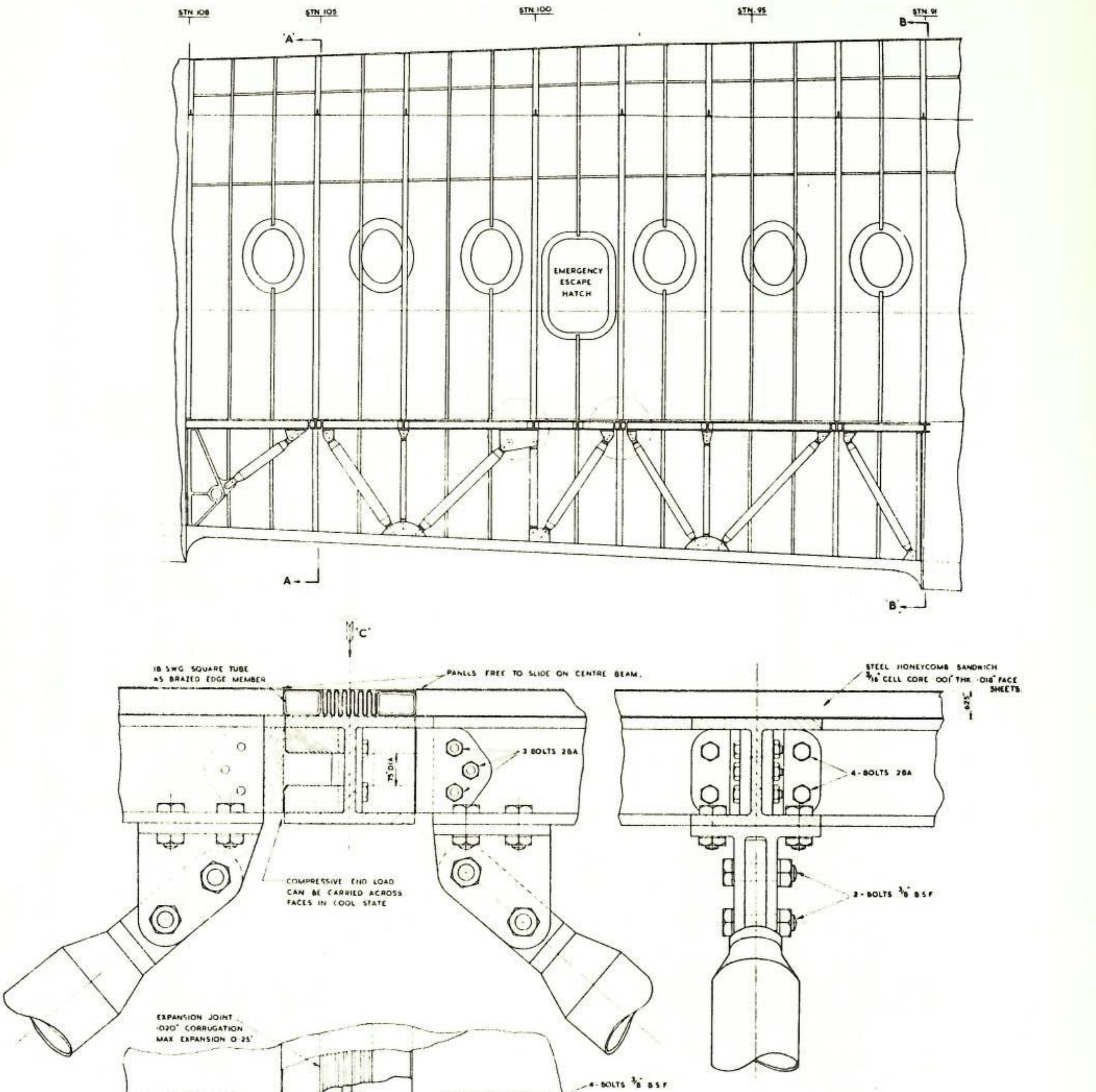


FIG.21. CABIN FLOOR DETAILS OF M=3·O A-62 DESIGN.

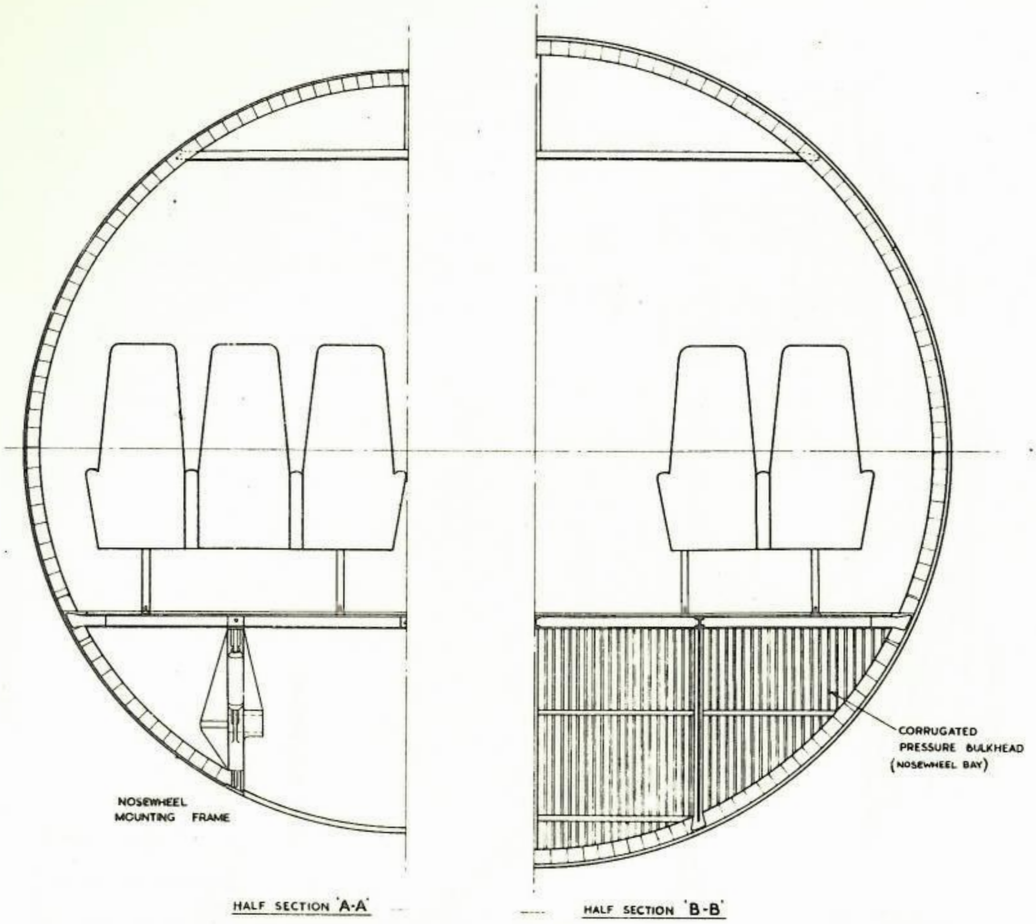


FIG.22. FUSELAGE BULKHEAD DETAILS IN NOSEWHEEL REGION OF M=3·O A-62 DESIGN.

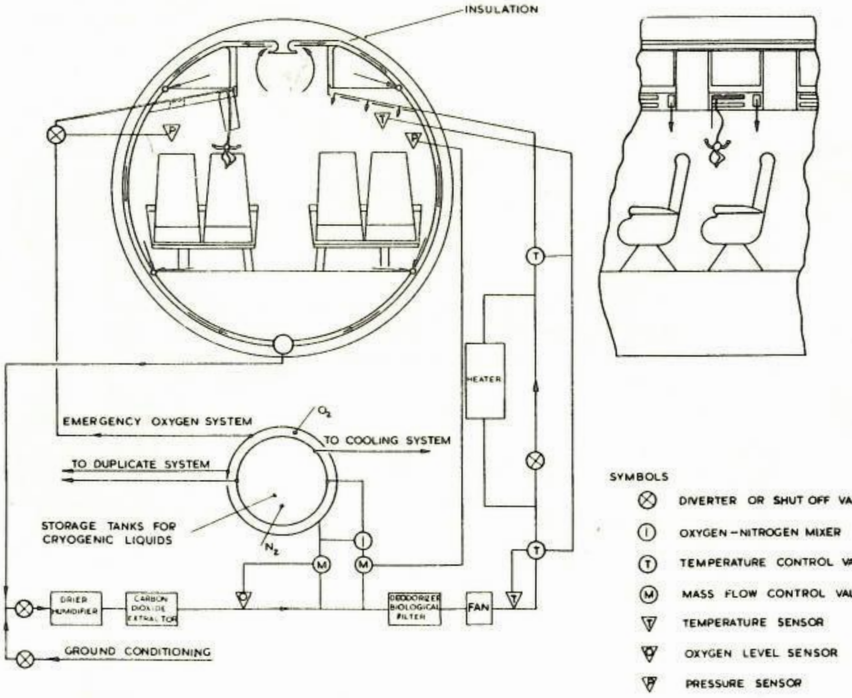


FIG.23. CABIN ENVIRONMENTAL CONTROL SYSTEM OF M=3·O A-62 DESIGN.

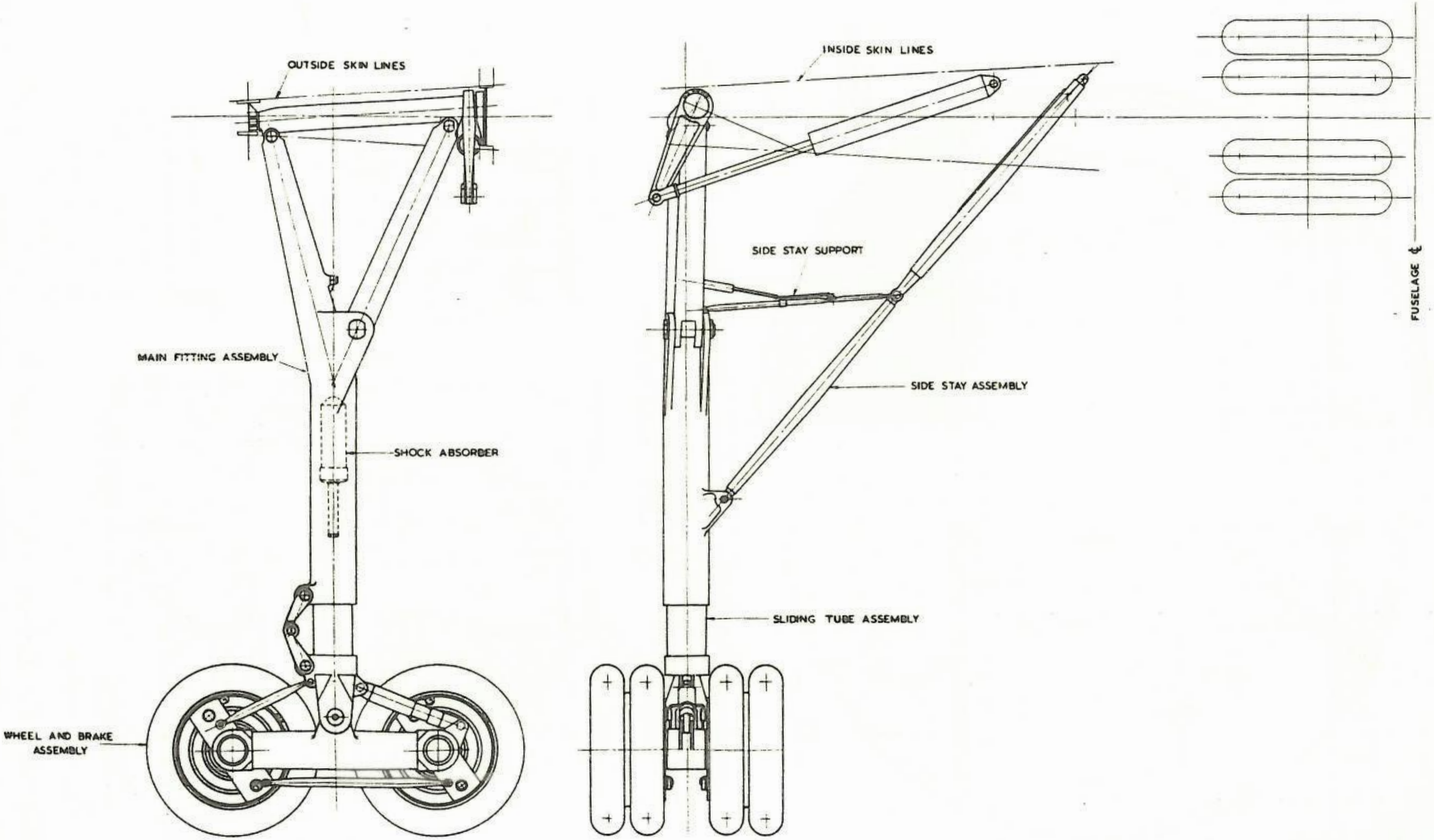


FIG. 24. ARRANGEMENT OF MAIN UNDERCARRIAGE OF M=3·0 A-62 DESIGN.

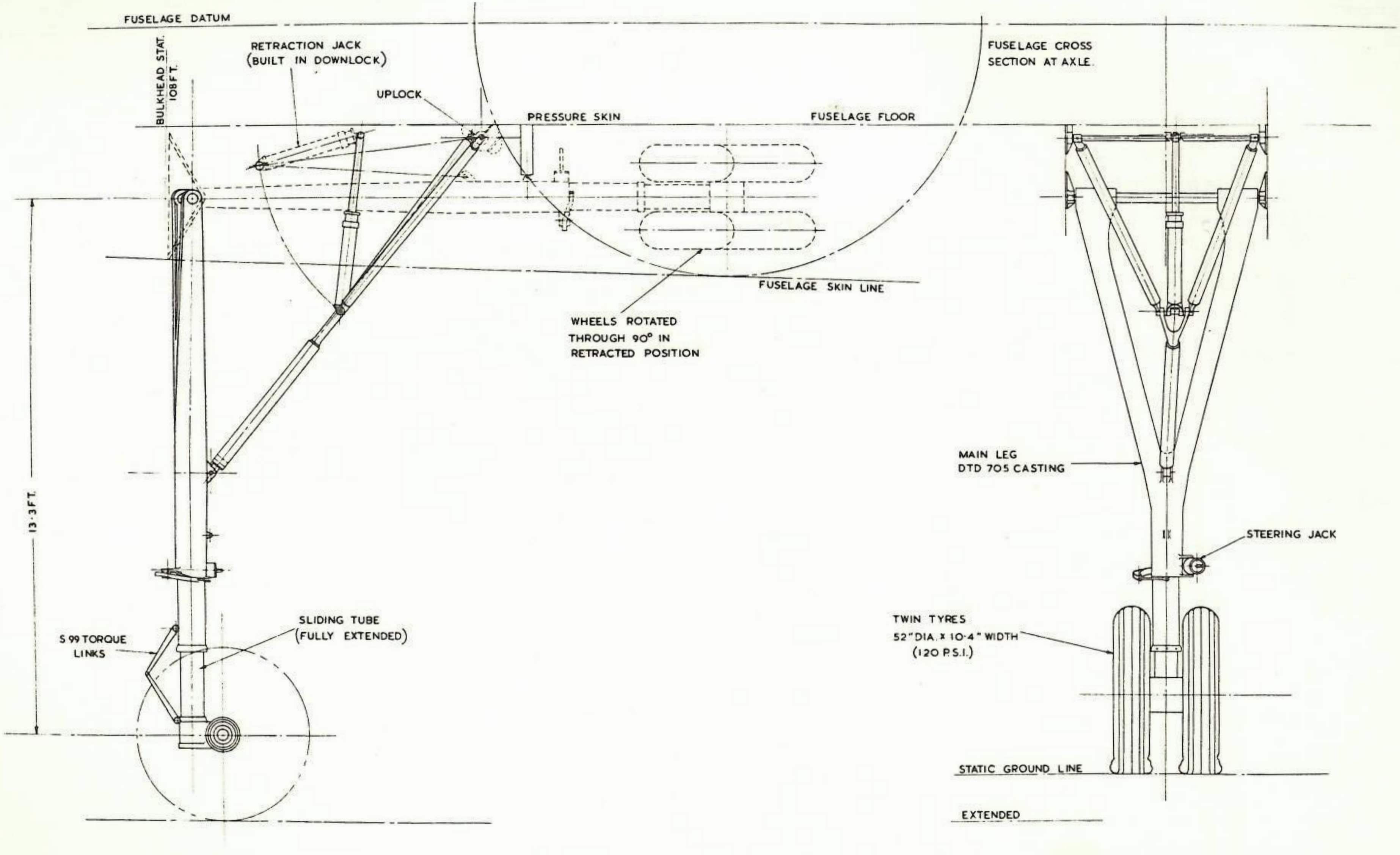


FIG.25. ARRANGEMENT OF NOSE UNDERCARRIAGE OF M=3·O A-62 DESIGN

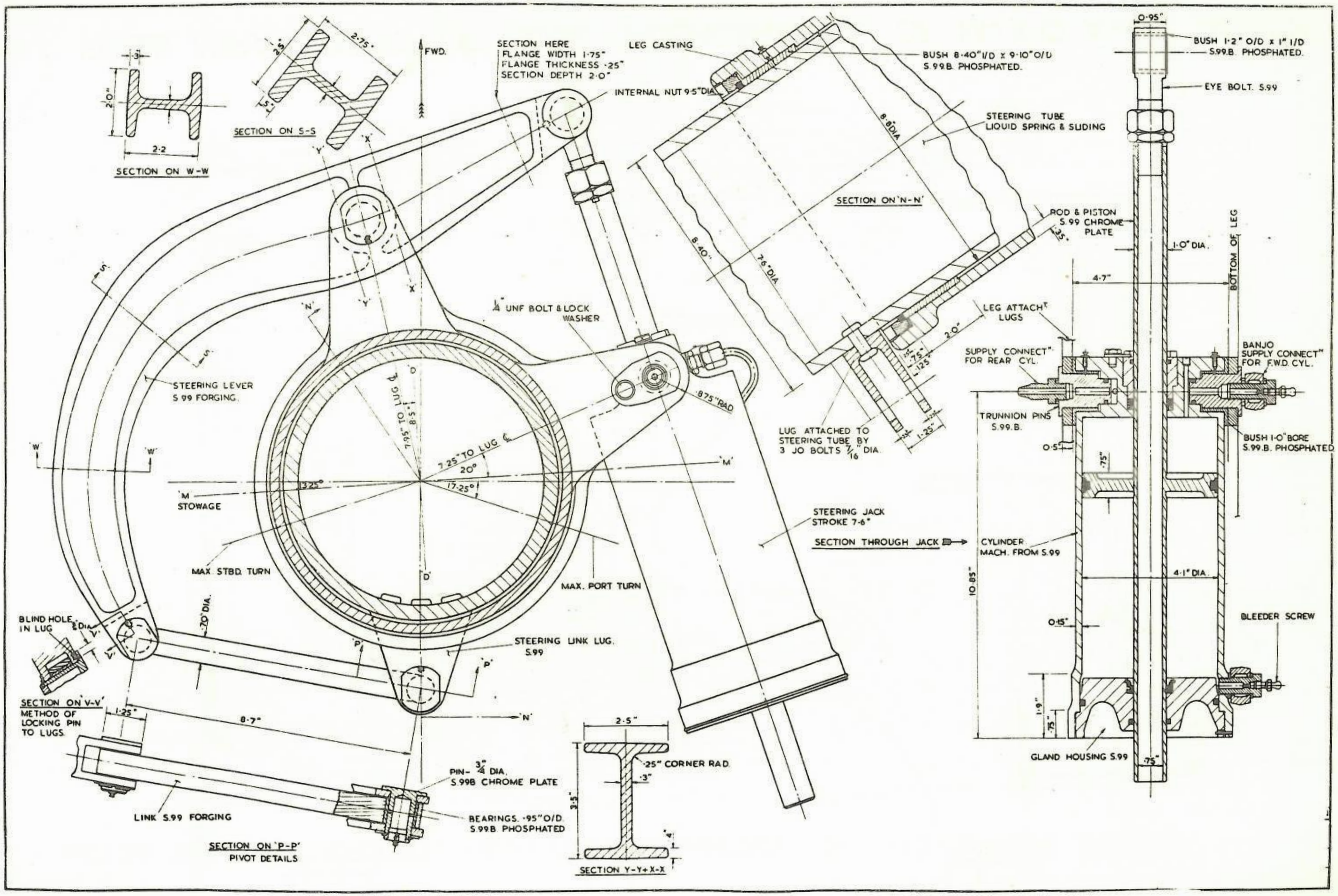


FIG.26. NOSEWHEEL STEERING DETAILS OF M=3 A-62 DESIGN.